Chapter 5
Aircraft Engines and Propulsion

Even considering the improvement possible, the gasturbine could hardly be considered a feasible application to airplanes, mainly because of the difficulty with the stringent weight requirements.

Gasturbine Committee, US National Academy of Sciences (1940)

It is gratifying to see the progressive clarification of ideas on the functioning of a simple device like a propeller, from the analogy with a screw jack to the complete theory based on the principles of fluid mechanics and using all the mathematical methods of this science.

Theodore von Kármán (1954)

There is a tendency in this age of high-speed aircraft to regard the reciprocating engine merely as an interesting holdover from the horse and buggy era of aviation. Yet in 1970, when over 105,000 aircraft will be operating in the category of general aviation, 99.7% of these airplanes will still be powered by piston engines.

C.N. Van Deventer (1965)

However great the progress in aerodynamic and structural efficiency has been over the years, the advances in light, compact and efficient powerplants have been paramount for the growth of fighter performance.

Ray Whitford [24] (1999)

5.1 History of engine development

All types of aircraft and rocket propulsion are based on the *reaction principle* derived from Newton's third law of motion: *To every action there is*

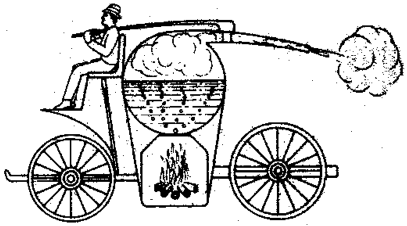


Figure 5.1 's Gravesande's "horseless carriage" with steam-jet propulsion (1680).

an equal opposed reaction.[1] The first attempts to use the reaction principle for rocket propulsion by means of black powder are usually attributed to China in the 13th century.[2] The general reaction principle was demonstrated by *Isaac Newton* in 1680 when he heated a gas and blew it off as a jet and a reaction force was measured in the opposite direction. Newton proposed an application of this principle to vehicle propulsion in the form of a steam wagon (Figure 5.1), which was to be demonstrated by the Dutch physician Willem Jacob 's Gravesande (1688–1742). The experiment failed because the steam-producing boiler was too heavy.

Aircraft rely on the reaction principle much more than ground vehicles. They almost exclusively use chemical energy released by combustion of liquid fuels or propellants. *Air breathing engines* use atmospheric air for generating power by combustion and obtain their *thrust* by reaction to the backward acceleration of ambient air or exhaust gases. Rocket engines are non-airbreathers carrying their *propellant* internally; they will only be mentioned superficially in this tbook.

Piston engines for aviation

First piston engines

In the 19th century the lack of suitable engines prevented aviation from making significant progress. This changed when the principle of the four-stroke *piston engine* cycle was invented by Beau de Rochas and the German

[1] In more precise terms this reads as follows: For every force acting on a body, the body exerts a force of equal magnitude in the opposite direction along the same line of action as the original force.

[2] According to credible sources, there was a fatal explosion of a manned rocket sleigh in 1232. Not until after the Second World War was rocket technology developed enough for manned space flight to be considered, which happened shortly thereafter.

N.A. Otto first built such an engine, from then on called an *Otto engine*. In contradiction to, for example, the steam engine, combustion of the rapidly burning gasoline fuel takes place inside the cylinders, viz. *internal combustion* (IC) by spark ignition (SI) of a fuel-air mixture. The pistons make a reciprocating motion.

The first application of Otto engines was in Russian *airships*, among others in O.S. Kostovic's Rossiya (1887). This engine had eight water-cooled horizontally opposed cylinders and delivered 60 kW of power at a mass of 240 kg – a very low specific mass (4 kg/kW) at that time. The improvements to the Otto engine by G. Daimler (1883) and K. Benz (1885) led to the beginning of the automobile industry, but also influenced aviation. Because the car engines at the beginning of the 20th century were too heavy for aircraft – they operated at 600–750 revolutions per minute (rpm) – pioneers like S.P. Langley and the Wright brothers had to design and build their own engines and propellers. That started the age of practical aircraft propulsion, which until the Second World War only consisted of the piston engine and the propeller. Until this time, piston engines for aviation are built as a unique category – apart from some types modified for ultra light aircraft, car engines are rarely installed in aircraft.

A remarkably sophisticated water-cooled petrol-fuelled engine with five radially placed cylinders was built by *C.M. Manly*, the assistant to *S.P. Langley*. With a basic mass of 57 kg, it created no less than 39 kW shaft power at 950 rpm (specific mass: 1.46 kg/kW).[3] The first piston engines built in large numbers were water-cooled *in-line engines* that had cylinders placed behind each other, needing bulky and draggy cooling radiators. The four-stroke piston engine developed by C. Taylor, assistant to the Wrights, falls into this category. At first this rudimentary but reliable 3.5 litre four-cylinder in-line gasoline-fueled engine generated about 9 kW of shaft power at 850 rpm. It was geared-down ($\approx$ 2:1) to a pair of 2.6 m diameter pusher propellers on either side of the wing, rotating in the opposite directions. The engine dry mass was 79 kg, including the radiator and piping. An improved version generated 22.5 kW at 1,300 rpm, with a mass of 99 kg (4.4 kg/kW). Although at the time these achievements were quite good, the more powerful engines that were developed later on by companies dedicated to aviation engine production had considerably better specific power and lower specific mass.

[3] With this achievement Manly deserved a better fate than his failed flying attempts with Langley's aircraft; see Section 1.5.

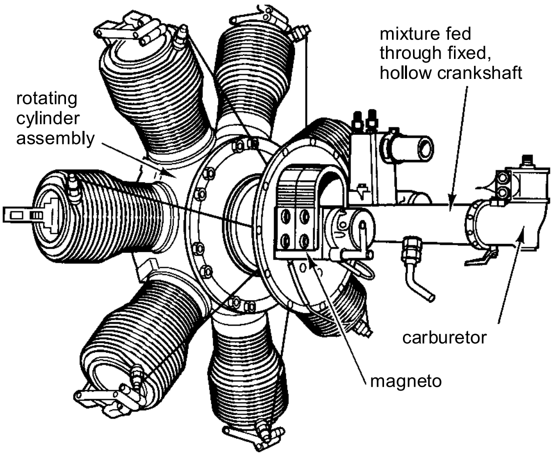


Figure 5.2 The Gnôme Omega seven-cylinder air-cooled *rotary engine* (1909), achieving 37 kW at 1,200 rpm [2, fig. 1-2-3].

Rotary engines

The *rotary engine* was invented by *L. Hargrave* in 1889. Air-cooled aircraft engines are derived from the French Gnôme five-cylinder rotary engine (1908). Its cylinders were radially oriented about the axis of rotation (Figure 5.2). The combination of crankcase, cylinders, and propeller rotated around the hollow crankshaft, which was attached to the aircraft. This ensured adequate cooling, even while running stationary or at low airspeeds. The crankshaft also served to supply the fuel-air mixture. These types of engines, then with two rows of cylinders, eventually developed 52 to 60 kW at 100 kg, less than 2 kg/kW.

Rotary engines were a significant improvement to the water-cooled types available at the time, but with badly controllable power output they had a high fuel and castor oil consumption and demanded a lot of maintenance. In the development of more powerful engines – with 9, 11 or 14 cylinders, a power output of 60 to 150 kW at about 1.35 kg/kW was achieved — the gyroscopic reaction of the rotating engine mass proved an insurmountable problem when manoeuvering with the small and light aircraft of that time. By 1920, design had reverted to a stationary engine secured to the fuselage, with a rotating crankshaft driving the propeller.

Progressing engine development

Between 1920 and 1945 there was an increasing demand for more powerful engines and, because of the limited power per cylinder, the number of cylinders per engine increased continuously. Amongst others, the Schneider Cup races for seaplanes was a strong drive for engine development. Piston engine development was divided between *air-cooled* and *liquid-cooled* gasoline-fuelled SI type engines. In air-cooled engines the cylinders were placed radially around the crankshaft to achieve maximum cooling. At first the *radial engine* and its cooling fins was completely exposed to the airflow, which caused a high drag. It was not until 1930 that a device was invented to reduce this – at first the primitive Townend ring around the cylinder heads, later the *NACA cowling*. This cambered cowling with cooling-air control flaps at the rear end much improved the external and internal flows, resulting in considerably reduced air drag and improved cooling. NACA engine cowlings were used in the Douglas DC-3 and thereafter in almost all civil aircraft with radial engines. Cooling was further improved with fans.

For high-speed application, the air-cooled radials, although lighter and more reliable, could not compete with liquid-cooled *in-line engines*. These had their cylinders placed in line with the longitudinal axis of the plane, enabling a compact installation with reduced air drag. During the period 1925–1935, the cooling radiators became more efficient and their drag decreased. A mixture of water and ethylene glycol was used for cooling, allowing operating temperatures of 140°C. At high altitudes, the engine power was increased by means of one or more *superchargers*. A supercharger is a compressor that boosts pressure and density of the *intake airflow*. This prevents the power from decreasing with altitude[4] from the level achieved at sea level because of the decreasing *air density*, an important asset for fighter aircraft and aircraft with a *pressure cabin*. The supercharger is driven mechanically by the engine shaft or aerodynamically by a turbine in the exhaust. Weighing air-cooled against liquid-cooled techniques usually meant that civil aircraft, bomber aircraft, recreational and training aircraft used air-cooled radial engines. Most racing and fighter aircraft used liquid-cooled in-line engines; after 1945 these were only built in small numbers.

Developments in metallurgy eventually enabled engines to run at 3,000–3,500 rpm. This strongly increased the power per litre of displacement (swept cylinder volume), but at such high rotational speeds, the propeller tip speed

[4] In the low levels of the atmosphere, ambient density falls by about one-half for every 6.5 km increase in altitude; see Section 2.6.

can become supersonic. This causes the propeller to have a low efficiency and creates objectionable noise. Instead of directly driving the propeller, high-speed engines reduce their output shaft speed by means of gearing, that is, a system of cogs between the crankshaft and the propeller. Although reduction gears have their mechanical complications and add extra weight and costs, they were – and still are – used in many piston engine types.

Another improvement in piston engine performance was possible due to the development of gasoline fuels with a high octane rating. This prevents detonation in the pistons and enables a higher *compression ratio*. Test flights with a Boeing YP-29 fighter aircraft (1934) showed that an increase in octane rating from 55–60 to 98 increased the maximum flight speed by 10% and the climb rate by 40%, while at the same time the take-off distance decreased by 30%. As of 1938, fuel with an octane rating of 100 was the standard for American fighter aircraft.

From 1930 onwards, the Junkers firm developed *diesel engines* for use in aviation. These are two-stroke opposed-piston water-cooled engines with a high compression ratio. This enables spontaneous ignition of the slowly burning diesel fuel and diesel engines are also called compression ignition (CI) piston engines. Although these were heavier than comparable Otto engines, their low fuel consumption made them attractive for use in long-distance aircraft. Only the 1940 Jumo 205 diesel engine (645 kW power at 2,800 rpm, mass 595 kg) was built in large numbers.

Propeller development

In the first decades of aviation only two-bladed wooden *fixed-pitch propellers* were used. The 1920s and 1930s saw improvements through the introduction of *adjustable-pitch* and *constant-speed propellers*. For high-power engines solid aluminum or hollow steel *propeller blades* were used instead of wooden blades. The constant-speed propeller ensures a constant engine shaft speed when varying the flight speed and therefore the maximum power is generated at a high *propeller efficiency*. Also the possibility to reverse the blade pitch allowed for a breaking force when *landing*. As the engine power increased, so did the propeller diameter and the number of blades, from two to four or more.[5]

[5] Propellers for the currently used gas-turbine engines are often completely or partly made of fiber-reinforced plastic and in order to reduce the noise produced, they rotate at a relatively low speed, using three to six blades.

Radial engines

Single-row *radial engines* were initially build with seven or nine cylinders, while the power increased from between 150 and 300 kW to about 900 kW. In the 1920s, a number of radial engines was built by the US engine factories Wright and Pratt & Whitney (P&W). The Wright Aeronautical Corporation, originating from the Wright brothers company, built the nine-cylinder 164 kW Wright J-5 Whirlwind with a mass of about 220 kg (specific mass: 1.34 kg/kW), that was installed in the Ryan Spirit of Saint Louis, the aircraft used by *Charles Lindbergh* in 1927 in his legendary transatlantic solo flight. Thereafter Wright developed a successful series of Cyclone engines, such as the R-1750[6] in 1927, improved in 1932 to the R-1820 and installed in the Douglas DC-3. Another version was tuned up to 937 kW at a specific mass of 0.68 kg/kW – comparable to the Rolls-Royce (R-R) Merlin liquid-cooled in-line engine – and 1,140 kW was achieved after 1945.

The first P&W radial engine, the R-1340 Wasp (1925), produced 317 kW at 1,900 rpm, with a specific mass of 0.93 kg/kW. In 1932, P&W went on to build radial engines with 14 cylinders placed in two rows of seven cylinders behind each other. Of these engines, the R-1830 Twin Wasp became one of the leading aircraft piston engines in history. Starting at 570 kW in 1933, the introduction of 100-octane fuel increased its power to 900 kW. The C-47 Dakota, the military version of the Douglas DC-3, was equipped with this engine. The English engine manufacturer Bristol developed a series of air-cooled engines which controlled the air and fuel supply and the dispatch of exhaust gasses by means of sleeve valves instead of poppet valves. One of these engines was the 14-cylinder Hercules, with a power of 1,200 to 1,500 kW. In Germany at the end of the 1930s, BMW built the formidable air-cooled 14 cylinder 41.8 litre BMW 801, initially with 1,175 kW take-off power and later tuned up to 1,470 kW.

Radial engines dominated in the bomber and transport aircraft operated during the Second World War. Although most fighter aircraft had in-line engines, there were important exceptions like the Republic P-47 Thunderbolt, the Focke Wulf FW 190 and Hawker aircraft (Typhoon, Tempest, Fury). After 1945, the development of radial engines for civil aircraft culminated in two-row 18 cylinder 2,100 to 2,800 kW engines and four-row 28 cylinder 2,600 to 3,400 kW engines. One example is the Wright Cyclone R-3500 18 cylinder engine, with 2,420 kW take-off power and 54.9 litre displace-

[6] This common US designation referred to the type (R for "radial") and the displacement, in this case 1,750 cubic inch (29 litre).

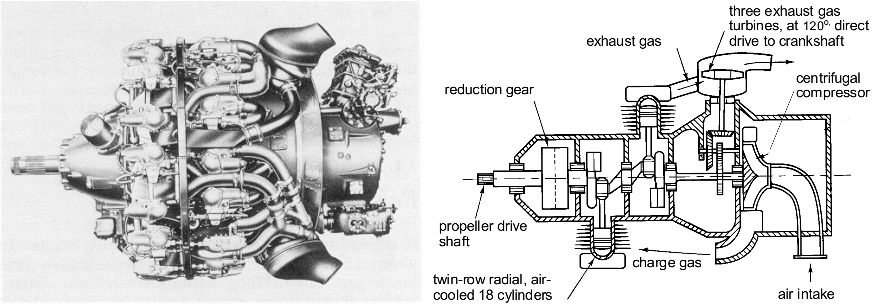


Figure 5.3 Wright turbo-compound 18R-3350: two-row, 18 cylinder, air-cooled, 54.8 litre, 2,625 kW [2, fig. 1-2-7].

ment. In the early 1950s this engine was introduced in the turbo-compound variety (Figure 5.3), with a power output of 2,625 to 2,750 kW at a specific mass of 0.60 kg/kW. Turbo-compound engines had turbines in the exhaust that delivered extra power to the engine shaft, which was also coupled to the supercharger. They were used, for instance, in the Douglas DC-7C and Lockheed Super Constellation, the last generation of long-distance civil aircraft to be equipped with piston engines. These had a low fuel consumption, ≈ 0.25 (kg/h)/kW, but when in service they proved to experience a lot of malfunctions, were unreliable and costly in maintenance. They proved to be the final stage in the development of the large radials and their production was terminated around 1960.

In-line engines

Liquid cooled in-line engines were further developed in many countries after the First World War, with the English R-R firm as one of the leading developers. The engines developed were used in great numbers in the allied fighter aircraft of the Second World War. The most important representative of these engines was the R-R Merlin, with two rows of six cylinders in a V-configuration (V-12), a displacement of 27 litre and cooled with a mixture of 30% ethylene-glycol and 70% water. Initially this engine, built in large numbers, had a power of 588 kW, later versions delivered 1,170 to 1,308 kW. The Merlin XX (Figure 5.4) used in the Hawker Hurricane delivered 926 kW at 2,850 rpm with a specific mass of 0.71 kg/kW. The Merlin 61 used a two-stage supercharger that raised the Spitfire's *ceiling* by 3,000 m and increased its maximum speed by 113 km/h. German counterparts of the Merlin were

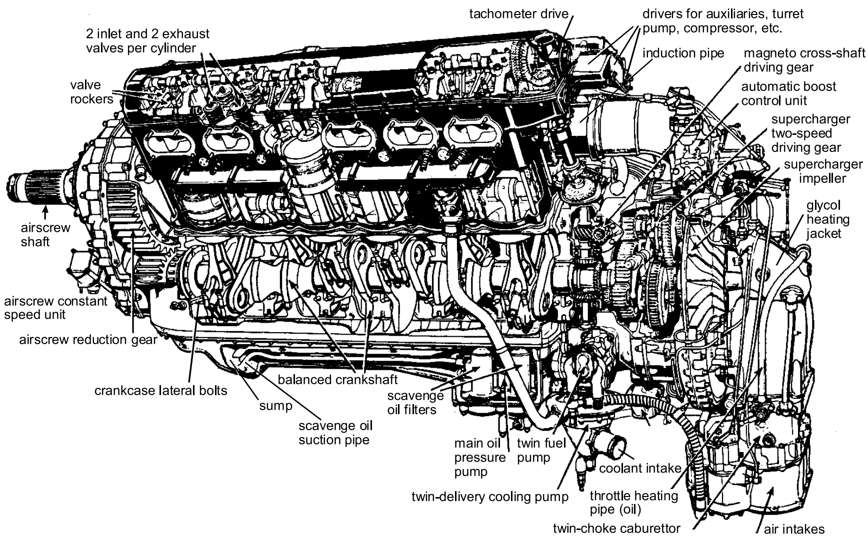


Figure 5.4 The illustrious Rolls-Royce Merlin XX, a liquid-cooled 27 litre V-12 in-line engine, equipped with a mechanically driven supercharger (courtesy of Rolls-Royce plc).

the Daimler-Benz DB 601 and the Junkers Jumo 211 and 213. However, they needed a larger displacement (about 35 litre) to generate the same power and were a lot heavier. On the other hand, their direct fuel injection was a distinct advantage in air combat, since the Merlin's carburettor often failed in manoeuvres with a negative *load factor*, such as nose-down *dives*. In-line engines developed later often had 24 cylinders in four rows of six cylinders each and delivered 1,200 to 1,800 kW. The 28 cylinder BMW-803 delivered a take-off power of 3,000 kW at a displacement of 84 litre and a specific mass of 0.87 kg/kW. Further development of large liquid-cooled engines stopped after 1945 since from then on fighter aircraft were equipped with jet engines.

Present-day piston engines

The piston engines used in civil aircraft in the 1950s marked the end of a long and intense development that reached to the boundaries of the mechanical and thermal loads. Although the large piston engines have all been replaced with gas turbine engines, small piston engines are still being used in the majority of light propeller aircraft. Most of these are derived from the class of Otto cycle engines from around 1932 with horizontally opposed air-cooled

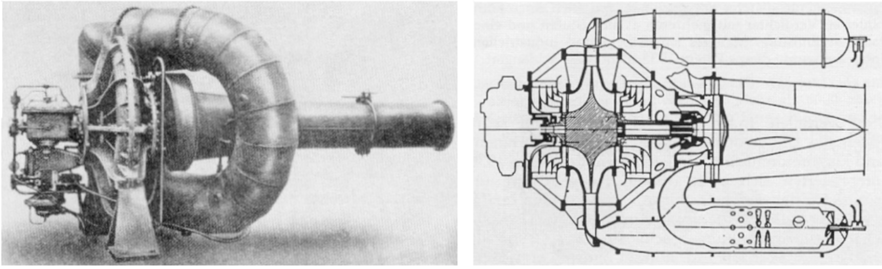
cylinders. After 1958, these flat (or boxer-type) engines were produced in large numbers by the manufacturers (Teledyne) Continental and (AVCO) Lycoming and in a modernized form they are still the standard for light aircraft. Most current flat engines have four or six (sometimes eight) cylinders, developing 75 to 300 kW power at a specific mass of 1.15 to 0.9 kg/kW, respectively. Some flat engines have liquid-cooled cylinder heads.

Another IC engine development is the four-stroke Wankel rotary. This is an RC engine that uses compression by means of a rotating instead of a reciprocating piston. The present-day renewed interest in the use of light diesel engines could lead to a threat for the SI piston engine. Moreover, traditional piston engines for general aviation now experience increasing competition from the *turboprop engines*. These are much lighter and do not need high-octane fuel since they run on the cheaper and widely available gas turbine fuel.

Development of gas turbine engines

Pioneers

The principle of the gas engine were already known in the 19th century – the Englishman John Barber obtained a patent on a turbine engine operating on gas in 1791 – but almost all developments were aimed toward delivering power to industrial (static) applications. The modern form of the turbine was used for the first time in a steam turbine by the Swedish engineer G. de Laval (1883). The Frenchman Maxime Guillaume obtained a patent on an aircraft gas turbine in 1922. It had a multistage axial supercharger, fuel injection, a combustion chamber, a multistage turbine and a starter motor. But only after experiments in the 1930s was the principle of jet propulsion successfully applied for the first time, in aviation applications in the Second World War. In a practical sense, gas turbine engines originate from the development of the *turbo-supercharger* for piston engines (1918) by the American *Stanley A. Moss*. In 1920, an aircraft with a turbocharger set an altitude record of about 10,000 m. Although the main parts of this supercharger are also important components of the *turbojet engine*, it was not Moss but the Englishman *Frank Whittle* and the German *Hans Pabst von Ohain* who are considered as the spiritual fathers of the aircraft gas turbine. Unaware of each other's research, they ran their first engines within a few weeks of one another in April 1937.



(a) Early design of Whittle's jet engine (1937) (b) Schematic cross section of the W2B, the first flyable jet engine (1942)

Figure 5.5 Milestones in the development of Whittle's jet engines with a two-sided centrifugal compressor.

In 1930, Whittle patented a turbojet engine with a multistage *axial-flow* and a *centrifugal compressor*, an annular combustion chamber, a single-stage axial turbine and a nozzle. The patent was awarded in 1932, but initially Whittle could not find the financial support to develop his design further. Therefore, he founded the Power Jets company together with former RAF pilots and started to build an experimental engine, the WU(1), in 1935. This engine (Figure 5.5a) had its first run on a test stand in April 1937 and was improved to become the WU(2) in 1938, which had a thrust of 2.1 kN.[7] It was only after this that the British Department of Aviation showed interest, followed by acknowledgement, subsidy, and eventually pressure to further develop the engine. In 1939 the construction of the Gloster E 28/39 was started, an experimental aircraft which had its first flight on May 15th, 1941 (Figure 1.23b). The Power Jets W1 engine installed in this aircraft weighed 2.8 kN and delivered a thrust of 3.8 kN,[8] giving the aircraft a speed of 483 km/h. A second aircraft had the W2B engine (Figure 5.5b) delivering 6.8 kN thrust and reached 750 km/h. However, the Gloster E 28/39 did not enter history as the first jet aircraft.

During his doctoral research in 1935, von Ohain had started to build a jet engine with a centrifugal compressor, combustion chamber and centripetal turbine. As of 1936 von Ohain was supported financially by the aircraft man-

[7] Initially the thrust of Whittle's engines were approximately as large as their weight. When an official questioned the progress of the project, Whittle answered: "Good progress has been made, Sir, except that the engine weight is what the thrust should be and the thrust is what the weight should be". In modern turbojet engines the maximum thrust is five to ten times the weight.

[8] If a thrust is mentioned, the static take-off thrust at sea level is meant.

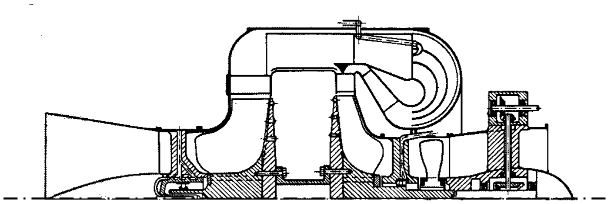


Figure 5.6 Schematic longitudinal cross section of von Ohain's engine, the He S 3B (1939) [6, p. 144].

ufacturer *Ernst Heinkel* (1888–1958). He tested a prototype engine 1937, the He S1, using hydrogen gas for fuel and delivering 2.5 kN of thrust. Simultaneously with further development of the engine, Heinkel built an experimental jet aircraft, the He 178. The configuration of the He S3B (Figure 5.6) installed in this aircraft had been inspired by a patent of Max Hahn and included a single-stage axial-flow compressor and a centrifugal compressor. Running on liquid fuel, it delivered 4.9 kN thrust at 3.6 kN weight. On August 27th, 1939, the He 178 (Figure 1.23a) became the first jet aircraft in history to take to the air. With an improved engine, the He S6 with 5.8 kN thrust, reached a speed of 700 km/h.

First operational jet engines

During WWII, Germany was extremely active in further developing gas turbine engines. The Junkers Jumo 004 jet engine with axial-flow compressor was first tested in 1940, the 1943 version Jumo 004B had a thrust of 8.9 kN and 6,000 Jumo 004A engines were built up to 1945. These were used, for instance, in the Messerschmidt Me 262 – the first operational jet aircraft – introduced by the Luftwaffe in 1944. Although these German jet aircraft, flying at more than 800 km/h, made a big impression on allied pilots, they had hardly any impact on the course of the war. During the war, Germany also worked on turboprop, pulse jet, and rocket engines.

Britain's Royal Air Force's first jet aircraft came into service in 1944. These were the Gloster Meteor 1, with two R-R W 2B Welland engines of 7.6 kN of thrust each. In 1943, R-R had taken over the responsibility of developing the engines from Power Jets. At the end of 1945, the version with a Derwent 8 engine reached a speed of 975 km/h.

In the United States, the General Electric Company (GE) was given the task in 1943 of developing jet engines for the US Air Force, in order to pro-

duce the Whittle type W 2B. This would lead to the GE J33 jet engine with a centrifugal compressor. The J33 engine was used in the first American jet aircraft, the twin-engined Bell XP-59A Airacomet, that made its first flight in October 1942. The first jet aircraft to be used by the USAF was the Lockheed F-80 Shooting Star that made its first test flights in January 1944. A second prototype of the F-80 had a 17.8 kN J33 engine. In 1949, the GE J47 became the first jet engine in America to be certified for civil aviation.

Turbojet and turbofan engines

The development of jet propulsion started with straight (or simple) *jet engines*. All air flowing through the *intake* of a straight jet engine is compressed to high pressure and then heated by combustion of kerosene or similar aviation fuel in the combustion chamber. After expanding in the turbine, which drives the compressor, the hot gas leaves the engine through the *exhaust nozzle* as a hight-speed jet. The turbojet cycle forms the basic element of all gas turbine engines in which hot gas is produced with energy that can be used for propulsion. In a straight turbojet nozzle the available kinetic energy is converted into jet thrust power, while the thermal energy is lost in the atmosphere.

Straight jet engines were introduced in civil aviation between 1950 and 1960, but soon had to make way for *turbofan engines*, or turbofans for short. In the dominant turbofan configuration, the intake air is first compressed in a low-pressure compressor or fan and thereafter split into two separate portions. The inner airflow enters the compressor(s) of a *gas generator*, also denoted as the *core engine*. This complicated high pressure engine section works similarly to a straight jet engine, except that a large amount of the available turbine power is used to drive the fan. The gas generator produces hot gas that is ejected through the nozzle at a much lower speed than the straight jet efflux speed. The outer fan airflow by-passes the gas generator through a duct and may or may not be mixed with the hot gas before leaving the engine nozzle. The average exhaust velocity of the jet of a turbofan engine is lower than that of a pure jet engine, which means that less fuel is consumed and less noise is produced by the exhaust. Turbofan engines are characterized by their *by-pass ratio*, that is, the ratio between the amounts of air by-passing and entering the gas generator. The first generation of turbofans used in civil aviation were *by-pass engines* with a multi-stage low-pressure compressor and a by-pass ratio between 0.2 and 1.0. The P&W JT3D and JT8D belong to this category as well as the R-R Conway, that had a thrust

of 78 kN at a specific weight[9] of 0.26. Straight jets and by-pass engines are no longer installed in civil aircraft because of restrictive noise regulations. Fighter aircraft however usually feature the more compact by-pass engines, often with *reheat*, also known as *afterburning* (Chapter 9).

Wide body airliners were introduced into service around 1970. They used turbofan engines, where the intake air is compressed by a *fan*, that is, a low-pressure compressor with a single row of blades and a relatively large diameter. The first generation of large turbofan engines – the P&W JT9D, the GE CF6, and the R-R RB 211 – were turbofans with by-pass ratios of up to five and a thrust of 180 to 250 kN at a specific weight of 0.18 to 0.20. They had a distinctively low fuel consumption and in spite of their high thrust, they produced considerably less noise than by-pass engines. These advantages are important enough for civil aircraft to make up for the disadvantages of higher complexity, purchase price and maintenance costs. In the 1980s and 1990s the large turbofans (R-R Trent, P&W 4000, GE 90) were improved versions of earlier types, with increased by-pass ratios up to 6–9 and take-off thrust up to 450 kN. The ongoing improvement of turbofan engines has made a significant contribution towards making the modern airliner economical, comfortable, reliable, and environmentally friendly.

Turboprop and turboshaft engines

A *turboprop engine* uses most of the power produced by the gas generator for driving the propeller shaft with a large step-down in speed by a reduction gear, the jet produces a relatively low thrust. The first turboprop engine to be built in large numbers was the R-R Dart that made its first test flight in October 1947. Several versions of this engine (Figure 5.7) were used in, for instance, the four engined Vickers Viscount and the twin-engined Fokker F27. The power was 738 kW at first and later increased to 1,550 kW. It was not long before turboprop engines generated more power than piston engines – between 1950 and 1953 the Kuznetsov NK12 was developed in the former Soviet Union with a formidable 10,300 kW of power – at only one-third of their specific mass. A typical example for the largest historical turboprops was the R-R Tyne, having a shaft power of 4,550 kW at a specific mass of 0.22 kg/kW. Nowadays, turboprop engines are available with power outputs between 250 and more than 10,000 kW. These are compact and reliable engines with a low specific weight and fuel consumption – the latter some-

[9] Instead of the specific mass of a propeller engine, we prefer to use the dimensionless ratio of weight to thrust for jet and turbofan engines.

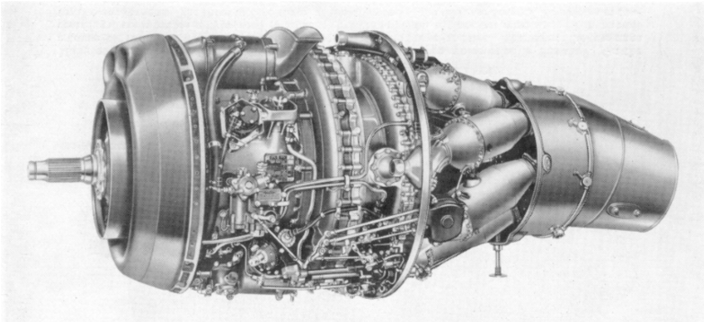


Figure 5.7 The RDa.3 with 1,045 kW take-off power was the first version of the Rolls-Royce Dart turboprop engine to be produced in series (courtesy of Rolls-Royce plc).

what higher than that of a piston engine with the same power – that are used mainly for regional airliners and general aviation.

Helicopters use *turboshaft engines* which convert all available turbine power for driving the rotor shaft and on-board systems. The exhaust gases expand to approximately ambient pressure. The first turboshaft engine-powered helicopter was the Alouette, which had a 1951 Turboméca Artouste installed, with 300 kW of power.

Recent developments

The gas turbine incited a revolution in aviation after the Second World War. Thanks to jet (and rocket) propulsion systems, installed in aircraft of revolutionary design, *supersonic speeds* were achieved within a few years. Gas turbine engines made it possible to achieve the high performance level of modern fighter aircraft and cleared the way for high-speed transportation with jet airliners.

Further developments in aircraft engines will play a key role in aviation, such as engines that combine the advantages of turbofan and turboprop engines to obtain more economical and environmentally friendly operations. A potential new generation of supersonic civil aircraft will make special demands on engine technology. The depletion of fossil fuels may lead to the use of synthetic and bio-fuels. Using hydrogen as a fuel while retaining the gas turbine technology is another option. However, in its liquid form LH_2 this cryogenic fuel requires radical changes to an aircraft's general arrangement and fuel system. In spite of the larger energy content per unit mass of LH_2,

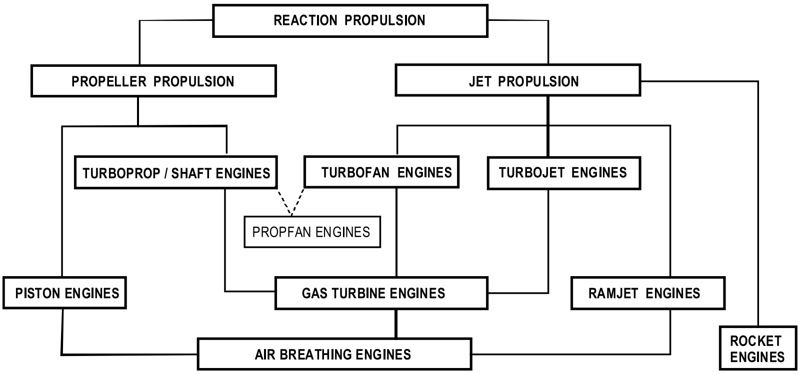


Figure 5.8 Classification of engine concepts mostly used in aviation.

its lower density when compared to kerosene requires a tank capacity that is four times larger for flying the same distance. Furthermore, the production and distribution will require the development of new infrastructure.

5.2 Fundamentals of reaction propulsion

Engine concepts

Figure 5.8 shows a schematic overview of the most important engine concepts used in aviation. Except the rocket engine, all of them are *air breathing engines* that can be used for atmospheric flight only. The following categories are distinguished:

- *Piston engines.*
- *Turboprop* and *turboshaft engines.*
- *Turbojet engines*, also called straight or simple jet engines.
- *Turbofan engines.*
- *Ramjet engines.*

This scheme is not only based on mechanical differences between propeller and jet propulsion, but also the energy conversion process has been taken into account. The different types will be discussed in the following sections, which will also discuss the technological implementation of the first four categories mentioned above. Gas turbine engines with reheat and ramjet engines will be discussed in Chapter 9. Besides the engine types mentioned in

Figure 5.8, there are several hybrids like the (now obsolete) compound diesel engine, the incidentally used turbo-ramjet, and the experimental ram-rocket.

The velocity increment of air needed for aircraft propulsion can be obtained in several ways.

- For *propeller propulsion* the air is accelerated backward with respect to the propeller plane using rotating *propeller blades*. The air force on the blades has a component in the direction of flight, the propeller *thrust*, and a component in the propeller plane responsible for its *torque*. Under a typical flight condition, this torque is compensated for by the torque of the piston engine or turboprop output shaft.
- For *jet propulsion* the gas generator produces a stream of hot gas through combustion at a near-constant (high) pressure. In a jet engine, the exhaust gas expelled from the turbine expands in the *nozzle* where the velocity increases to form the *propulsive jet*. In turbofan engines there is, in addition, a cold airflow that is accelerated by a (low pressure) compressor or *fan*. After by-passing the gas generator, the cold flow expands in a secondary nozzle.

Although propeller and jet propulsion both use the reaction principle, there are some fundamental differences with respect to practical application. These will be explained hereafter by examining which factors influence the propulsive force.

Propeller thrust

A *propeller* is a device by which a small velocity increment is given to a large mass of fluid. It is merely a collection of rotating aerofoils using sections similar to those used on wings. A first approximation for *propeller thrust* is obtained by applying the momentum equation to the axial flow *actuator disc*. This concept was developed for ship propellers by the Englishman *W. Froude* (1810–1879) and the Scotsman *W.J.M. Rankine* (1820–1872). The Rankine–Froude theory can be used for all types of single-rotor propellers as normally used on aircraft, contra-rotating pairs of propellers, ducted rotors and helicopter rotors. In this model, the flow around the propeller blades is not examined in detail, rather the propeller is assumed to have a large number of blades so that it is replaced by an infinitely thin disc in its plane of rotation. The disc causes an instantaneous and uniform velocity and pressure increment of the air flowing through it; see Figure 5.9(a). The accelerated flow is called the *slipstream* (index s).

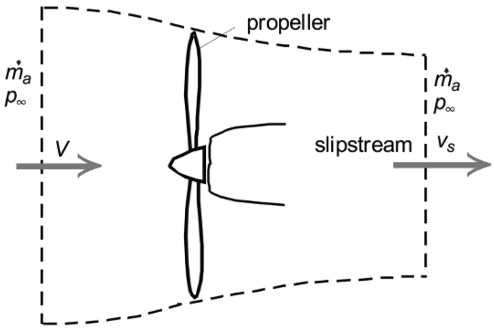
The actuator disc thrust follows from the *momentum equation* for *steady flow*, derived from Newton's second law; see Section 3.3 and Equation (3.26). The *control area* for the present application consists of the *stream tube* marked with a dotted line and the planes perpendicular to the flow far upstream and downstream. The air (index a) mass flow rate through the disc $\dot{m}_a$ is equal to the slipstream mass flow rate. The entry velocity far upstream is equal to the flight speed V, the fully developed slipstream velocity relative to the actuator disc is denoted v_s. In both surfaces the static pressure is equal to the ambient pressure p_∞. In the momentum equation R_x denotes the resultant pressure force in the direction of motion on the side walls of the *stream tube*. There is an equal force in the opposite direction on the surrounding flow. Since there is no change in velocity and therefore no change in momentum in this flow, there is also no force acting on the side wall of the stream tube: $R_x = 0$. The only force on the flow is a backward pressure force exerted by the disc leading to the momentum increase between the entry and the exit surfaces; the reaction to this is the disc thrust. Application of Equation (3.26) with $p_1 = p_2 = p_\infty$ yields

$$T = \dot{m}_a v_s - \dot{m}_a V = \dot{m}_a (v_s - V) \,, \tag{5.1}$$

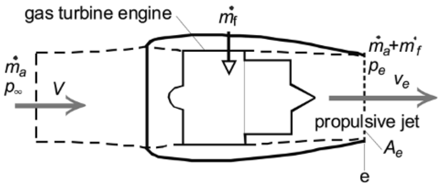
where the thrust T is positive in the forward direction. The terms $\dot{m}_a v_s$ and $\dot{m}_a V$ describe *momentum flow* rates through the stream tube and for $v_s > V$ the propeller experiences a positive thrust. The concept of the actuator disc will be discussed in detail in Section 5.9.

Turbofan thrust

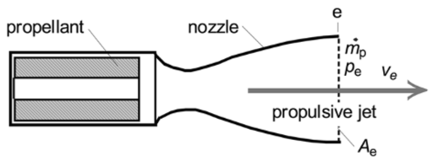
The *intake* air of a turbofan, see Figure 5.9(b), has a mass flow $\dot{m}_a$ and upstream flow conditions are equal to the flight speed V and the ambient pressure p_∞. The efflux through the nozzle exit (index e) with area A_e has a supposedly uniform velocity v_e relative to the engine and static pressure p_e. The fuel (index f) mass flow rate $\dot{m}_f$ into the combustion chamber is assumed to be equal to the mass flow rate of the combustion products, leaving the engine with velocity v_e. The control area for applying the momentum equation (3.26) consists of the *stream tube* marked with a dotted line, the plane far upstream of the intake and the nozzle exit. Similar to the actuator disc treated above, the resultant pressure force on the outer flow is equal to



(a) Airflow through a propeller



(b) Airflow through a jet engine



(c) Principle of a solid rocket engine

Figure 5.9 Derivation of the thrust from the momentum equation.

zero,[10] hence $R_x = 0$. The *thrust* follows from

$$\begin{aligned} T &= (\dot{m}_a + \dot{m}_f)v_e - \dot{m}_a V + (p_e - p_\infty)A_e \\ &= \dot{m}_a\{(1+f)v_e - V\} + (p_e - p_\infty)A_e, \end{aligned} \tag{5.2}$$

where $f = \dot{m}_f/\dot{m}_a$ denotes the fuel/air mass flow ratio. In a jet engine this ratio is $f \approx 0.015$ to 0.020, in *turbofan engines* it is even lower and therefore we may assume $1 + f \approx 1$. Moreover, since the pressure term in Equation (5.2) is often much smaller than the momentum term, it will be neglected. Alternatively, if we use the average velocity of the fully expanded jet (index j) instead of the nozzle exit velocity v_e, the thrust equation becomes similar to the propeller thrust equation,

$$T = \dot{m}_a(v_j - V). \tag{5.3}$$

Strictly speaking, this result only holds if $p_e = p_\infty$ or if the exit plane of the control area is located downstream of the engine where the jet has expanded to p_∞. Since the exhaust jet has mixed with the surrounding air, the

[10] The friction force on the engine nacelle is taken into account as a drag component instead of a thrust loss.

velocity v_j is a fictitious quantity and for this reason the thrust according to Equation (5.3) is called the *ideal thrust*. Also it should be kept in mind that in a turbofan engine the hot jet velocity will be considerably higher than that of the by-pass air. If both jets are unmixed inside the engine, the momentum equation must be applied to each flow.[11] It is also obvious that Equation (5.3) holds for the ideal thrust of turbofans as well as for straight turbojets.

Equation (5.2) defines the force acting backwards on the airflow as a momentum term and a pressure term. According to the reaction principle, the thrust is acting in forward direction as pressure and friction on the internal engine surfaces and on the internal surface of the nacelle, that is, the air intake and the by-pass duct. It is transmitted to the aircraft by the engine attachment structure.

Rocket engines

A *rocket engine* is a device that burns fuel and an oxidizer, both of which are carried by the vehicle. The *propellants* of a rocket engine are either liquid or solid. Liquid propellant rockets employ liquids which are supplied under pressure from tanks to the combustion chamber. Solid propellant rockets contain the propellant within the combustion chamber, as depicted in Figure 5.9(c). After combustion of the propellants (index p) at a rate of $\dot{m}_p$, the combustion products are accelerated in the nozzle. Dependent on the exit area A_e they expand to pressure p_e, reaching a velocity v_e. According to the reaction principle, forward thrust is obtained in reaction to the rearward momentum of the combustion products and the overpressure at the nozzle exit,

$$T = \dot{m}_p v_e + (p_e - p_\infty) A_e. \tag{5.4}$$

The operation of a rocket engine is independent from the atmosphere and the flight speed, although thrust does depend on the ambient pressure. This type of engine can generate thrust in a vacuum and is therefore the pre-eminent means of propulsion for space flight. A rocket launcher used for starting the flight of a spacecraft from earth has to generate a very high thrust, so that a very large high-speed propellant mass flow is required. Therefore, rocket engines have an operating time of a few minutes and are rarely used in aviation.

[11] In the case of mixed exhaust streams, an additional solution is necessary for the mixing of both nozzle flows. For a mixed flow, the thrust is slightly higher than for separate flows.

Propulsive efficiency and specific thrust

For propeller propulsion, the engine power is conveyed to the propeller, delivering thrust because the slipstream has a higher velocity relative to the aircraft than the *undisturbed flow*. The kinetic energy per unit of mass of the slipstream is higher than that of the undisturbed flow. The conversion of shaft power delivered to the propeller into propulsive power is accompanied by a kinetic energy increment of the ambient air that cannot be regained and hence must be considered as a power loss. Similarly, for jet propulsion, the engine generates jet power that is converted into propulsive thrust power, also with a kinetic energy increment as a side-effect. This can be expressed as a *propulsive efficiency* loss, defining a significant contribution to the required fuel consumption. Propulsive efficiency is therefore a suitable criterion for comparing different types of aircraft propulsion.[12]

The jet power P_j is equal to the kinetic energy increment of the air per unit of time imparted by the propeller or the jet engine. If we use v_j to also denote the velocity of the expanded slipstream (Figures 5.9a and b), then for propeller and jet propulsion we find the same expression:

$$P_j = \frac{1}{2}\dot{m}_a(v_j^2 - V^2). \tag{5.5}$$

According to Equations (5.1) and (5.3), the thrust produces *available power*

$$P_{\text{av}} = TV = \dot{m}_a(v_j - V)V. \tag{5.6}$$

The *propulsive efficiency* is defined as

$$\eta_j \stackrel{\wedge}{=} \frac{P_{\text{av}}}{P_j} = \frac{(v_j - V)V}{\frac{1}{2}(v_j^2 - V^2)} = \frac{2V}{v_j + V} = \frac{2}{1 + v_j/V}\,. \tag{5.7}$$

This is also called the *Froude efficiency*, after the earlier-mentioned *W. Froude*. It can be related to the *specific thrust*, which is the propulsive force per unit of mass flowing through the stream tube,

$$\frac{T}{\dot{m}_a} = v_j - V. \tag{5.8}$$

The dimension of the specific thrust is $\text{N}(\text{kg/s})^{-1}$, equal to the dimension of speed. From Equations (5.7) and (5.8) it follows that

[12] The efficiency of aircraft propulsion can be compared with the fraction of the engine power of a ground vehicle that is available for overcoming its air drag, which is high. Ground-vehicle propulsion by means of ground friction is therefore more efficient than reaction propulsion of aircraft.

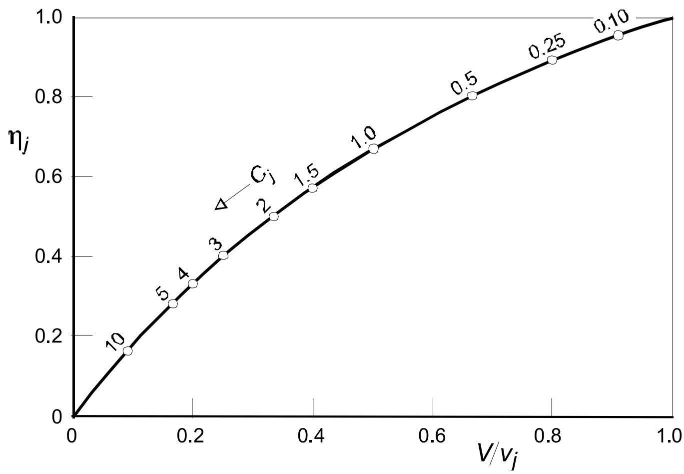


Figure 5.10 Relation between the *propulsive efficiency*, jet velocity coefficient and the ratio of flight speed to jet velocity.

$$\eta_j = \frac{2}{2 + T/(\dot{m}_a V)} = \frac{2}{2 + C_j} . \tag{5.9}$$

Here we have introduced the jet velocity coefficient

$$C_j \triangleq \frac{T}{\dot{m}_a V} = \frac{v_j}{V} - 1 , \tag{5.10}$$

which is the thrust delivered per unit of air *momentum flow* rate through the propeller or the jet engine. The propulsive efficiency increases when C_j and v_j/V are reduced in value. In Figure 5.10, the propulsive efficiency is given as a function of V/v_j. Also the corresponding C_j values have been given, with the following characteristic values:

- The static situation ($V = 0$), where $\eta_j = 0$. Since the propeller or jet engine delivers a propulsion force $T = \dot{m}_a v_j$, we have $C_j = \infty$.
- For $v_j = V$ we have $\eta_j = 1$, whereas the propulsive force is zero, so $C_j = 0$.

Table 5.1 forms another illustration, showing typical properties for different types of propulsion and flight conditions. From this table we can derive the following:

- The propeller has a high propulsive efficiency because the slipstream velocity is only slightly higher than the flight speed. Later on in this

Table 5.1 Example of *propulsive efficiency* data.

Type of propulsion	*Alti-tude* km	*Flight speed* V, m/s (Mach no.)	*Jet velocity* v_j, m/s	*Speed ratio* v_j/V	*Specific thrust* $T/\dot{m}_a$, m/s	*Jet coefficient* C_j	*Propulsive efficiency* η_j
propeller	6	150	160 (0.47)	1.07	10	0.067	0.97
subsonic jet engine	9	250 (0.82)	750	3.00	500	2.00	0.50
low BPR turbofan	9	250 (0.82)	582*	2.33	332	1.33	0.60
high BPR turbofan	9	250 (0.82)	418*	1.67	168	0.67	0.75
supersonic jet engine	16	600 (2.03)	1,000	1.67	400	0.67	0.75

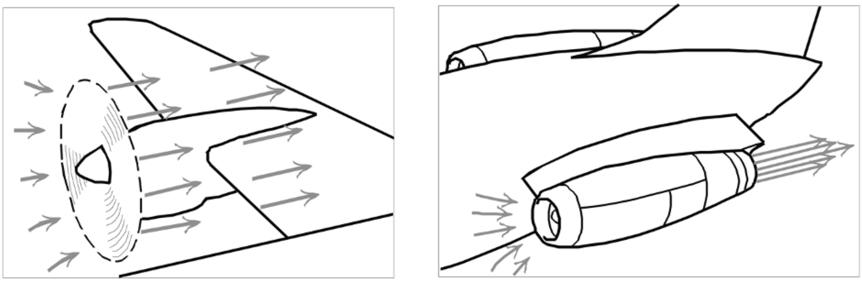
* weighted average of primary and secondary airflow

chapter it will be shown that in reality the *propeller efficiency* is lower because of additional losses.

- In a straight turbojet engine, all inlet air is used in combustion. Its relatively low mass flow incurs a much higher jet velocity than the flight speed, which makes the propulsive efficiency low.
- In the low by-pass ratio (BPR) turbofan the average jet velocity decreases since part of the inlet air does not participate in combustion, but is compressed and then expands. The propulsive efficiency of this engine is 20% higher than that of the turbojet.
- In the high by-pass ratio (BPR) turbofan the inlet air mass flow is considerably larger than that through the gas generator. The table shows that, for the same conditions, this gives it a 50% higher propulsive efficiency than the turbojet.
- The propulsive efficiency of a turbojet engine depends strongly on the flight speed. At *supersonic speed* it is much higher than at *subsonic speed*, equal to that of a high by-pass turbofan at transonic speed.

Characteristics of reaction propulsion

For a given flight speed, the propulsive efficiency decreases when the jet velocity increases. Although the specific thrust increases, the kinetic energy



(a) A propeller imparts a small velocity increment to a large mass of air

(b) A turbojet engine imparts a large velocity increment to a (relatively) small amount of air

Figure 5.11 The propulsive efficiency of a propeller is higher than that of a jet engine.

of the exhaust gasses increases quadratically. This energy is not used for propulsion and its only effect is the warming up of the atmosphere. Therefore, it is energetically more favourable to generate thrust with an engine type that expels its exhaust gasses at a relatively low velocity. If a propulsive efficiency of over 90% is to be reached, Figure 5.10 shows that the specific thrust must be less than 20% of the flight speed, which in practice is only possible for propeller aircraft. The difference between propeller and jet propulsion is clearly exemplified in Figure 5.11.

For a given jet velocity, the propulsive efficiency increases and the specific thrust decreases when the flight speed increases. Engines with a high jet velocity will therefore have a poor efficiency in low-speed flight, but may perform quite well at supersonic speeds. To obtain an acceptable propulsive efficiency for long-distance flights, a favourable ratio of specific thrust to flight speed must be sought. For instance, Figure 5.10 shows that for $\eta_j > 2/3$ it is required that $v_j < 2V$ and then the specific thrust is lower than the flight speed.

For a given jet or turbofan thrust, the engine weight decreases when the airflow decreases. Aiming at a light and compact engine then leads to a high specific thrust which, however, contradicts the conditions for a high propulsive efficiency to reduce fuel consumption. Especially when turbofan engines are used, a compromise has to be made between low fuel consumption and a light, compact engine. In civil aircraft the take-off thrust and the power plant weight are often less than 30 and 8%, respectively, of the aircraft weight and the decision is made in favour of turbofan engines which have a low specific cruise thrust and fuel consumption. Fighter aircraft, on the other hand, have a thrust of the same order of magnitude as the aircraft weight and, hence, these

aircraft have strict demands on engine compactness. A relatively small airflow producing a large thrust is thus required, typically leading to selection of a low BPR engine with *reheat*.

5.3 Engine efficiency and fuel consumption

Because engines are used in different applications, they often have very different performance levels. When the fundamentals of propulsion were discussed, the engine's internal processes were not taken into consideration.[13] Therefore the formulas found cannot be used to make a straightforward comparison between engine types or to calculate aircraft performances. However, the characteristic numbers that will be presented in this section are derived from actual engine performance specifications and they are representative of present-day engines and aircraft.

Total efficiency

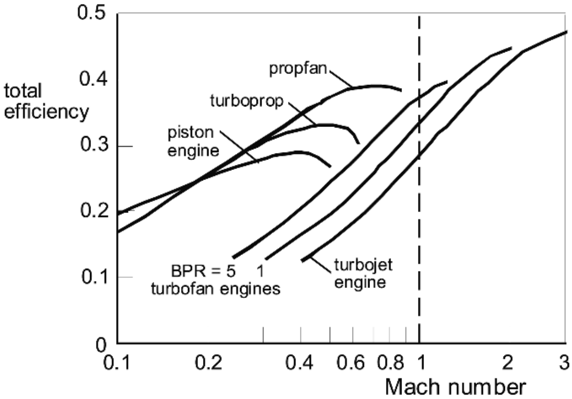
Although engines exhibit large variations in thrust, power, and fuel consumption, it makes sense to compare their *total efficiency*, also known as the overall efficiency. This is defined as the ratio of the available propulsive power P_{av} to the energy content of the fuel used to generate it,

$$\eta_{\text{tot}} \triangleq \frac{P_{\text{av}}}{\dot{m}_f H} = \frac{TV}{\dot{m}_f H} \,. \tag{5.11}$$

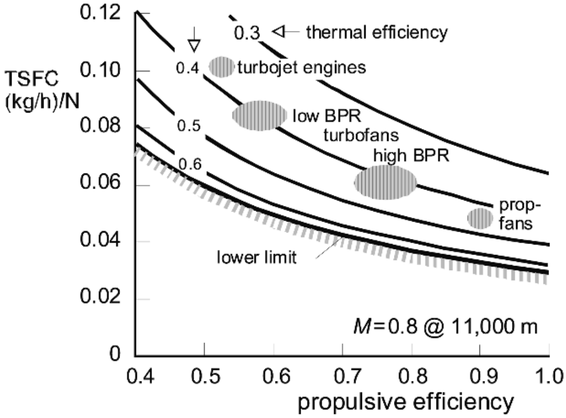
Here H is the *heating value* or calorific value of the fuel, which indicates the amount of heat that is released when the fuel is burnt completely. For hydrocarbon fuels this is $H \approx 42 \times 10^3$ kJ/kg. A typical turbofan total efficiency is between 0.30 and 0.35 for transonic cruising, at supersonic speed ($M = 2$) it can be between 0.40 and 0.45 (Figure 5.12a). Total efficiency is the product of the thermal and the propulsive efficiencies.[14] Using this subdivision, the influence of the flight speed on the total efficiency will be clarified.

[13] The analysis of internal engine processes is carried out in the form of cycle analysis based on thermodynamic principles, which is outside the scope of this book. An excellent introduction to this important subject can be found in [2].

[14] Strictly, the efficiency of the combustion process should also be taken into account, but for modern engines this is very near to 100%.



(a) Variation of total efficiency with flight speed



(b) Specific fuel consumption and efficiencies

Figure 5.12 Overview of global characteristics for different classes of aircraft propulsion.

Thermal efficiency

The *thermal efficiency* is the fraction of the fuel energy content that is converted into mechanical (shaft) power, or into the kinetic energy increment of the propulsive jet(s). The definitions of this efficiency for propellers and jet engines work out differently:

$$\text{propeller propulsion:} \quad \eta_{\text{th}} \triangleq \frac{P_{\text{br}}}{\dot{m}_f H}\,, \tag{5.12}$$

$$\text{jet propulsion:} \qquad \eta_{\text{th}} \stackrel{\wedge}{=} \frac{P_j}{\dot{m}_f H} = \frac{\frac{1}{2}\,\dot{m}_a(v_j^2 - V^2)}{\dot{m}_f H} \,. \tag{5.13}$$

In the first equation, P_{br} denotes the (brake) horsepower delivered to the propeller shaft, with the mechanical engine losses discounted. In the second equation, P_j denotes the excess kinetic energy of the jet relative to the intake air. For a given flight speed and altitude, the thermal efficiency is mainly determined by the thermodynamic cycle of the engine. For piston engines this depends mainly on the (volumetric) *compression ratio*, for gas turbine engines the *total pressure* and temperature at the combustion chamber exit relative to the ambient conditions are most important. Calculation of the thermal efficiency is too elaborate to fit within the scope of this book; the reader who is interested in cycle analysis is referred to the bibliography mentioned at the end of this chapter.

Propulsive and propeller efficiency

In Section 5.2 the *actuator disc* was used as an idealized model for the propeller. In reality the propeller thrust is less than the value given in Equation (5.1) since it is diminished by losses: profile drag of the blades, rotational energy in the slipstream and non-uniformity of the thrust distribution. The propulsive efficiency is therefore replaced by the *propeller efficiency*,

$$\eta_p \stackrel{\wedge}{=} \frac{P_{\text{av}}}{P_{\text{br}}} = \frac{TV}{P_{\text{br}}} \,, \tag{5.14}$$

For jet propulsion (index j) the *propulsive efficiency* is defined as

$$\eta_j = \frac{P_{\text{av}}}{P_j} = \frac{TV}{\frac{1}{2}\dot{m}_a(v_j^2 - V^2)} = \frac{2}{1 + v_j/V} \,. \tag{5.15}$$

It should be noted that a turbofan engine usually has two jets with different velocities, for which the average jet velocity v_j has to be used. Moreover, the actual gas generator power which is available for propulsion is sometimes used, instead of the ideal power P_j figuring in Equations (5.13) and (5.15).

The definitions of efficiencies have been chosen in such a way that for propeller propulsion $\eta_{\text{tot}} = \eta_{\text{th}}\eta_p$ and for jet propulsion $\eta_{\text{tot}} = \eta_{\text{th}}\eta_j$. In spite of differences in the partial efficiencies, their product – the total efficiency – is unambiguous and can be used to compare different types of engines. Figure 5.12b gives a global overview of the efficiencies of the most important gas turbine engines for aeronautical use.

Influence of the flight speed

Since the thermal efficiency of a piston engine is independent of the flight speed and that of a gas turbine engine varies only slightly, it is mainly the propulsive efficiency that determines the variation of total efficiency. In Figure 5.12(a), it is shown that for $M > 0.2$ the combination of turboprop and propeller has a slightly higher total efficiency than a piston engine with propeller. From $M \approx 0.6$ and higher, the propeller efficiency deteriorates because of *compressibility* effects. At low subsonic speeds, turbofan engines have a low efficiency and thus a high fuel consumption, at high subsonic speeds a high BPR turbofan efficiency equals or surpasses the efficiency of a turboprop. Straight turbojet engines are very inefficient at low speeds, but at $M = 2$ they achieve a high efficiency. This is mainly due to the high thermal efficiency, whereas the propulsive efficiency at $M = 2$ is comparable to that of a high BPR turbofan at $M \approx 0.8$ (see also Table 5.1).

Specific fuel consumption

Most efficiencies defined above are not commonly used in engineering practice. Instead, engine performance data are usually specified in terms of absolute values of thrust and fuel consumption, or expressed as *specific fuel consumption* (SFC). Fortunately, total efficiencies can readily be derived from these data and for several reasons this can be an appropriate action for the inexperienced analyst. The specific fuel consumption of an air breathing engine is defined as the ratio between the fuel consumption per unit of time F and the shaft power P_{br} or jet thrust T:

$$\text{brake specific fuel consumption (BSFC):} \quad C_P \triangleq \frac{F}{P_{\text{br}}} \,, \tag{5.16}$$

$$\text{thrust specific fuel consumption (TSFC):} \quad C_T \triangleq \frac{F}{T} \,. \tag{5.17}$$

For turboprops, the shaft power is often replaced with the *equivalent power*, which includes the power delivered by the engine exhaust (Section 5.7). Contrary to an efficiency, a *specific fuel consumption* is not a dimensionless quantity and its value depends on the system of units used. Furthermore, fuel consumption can be expressed as a mass flow rate ($F = \dot{m}_f$) or as a weight flow rate ($F = \dot{m}_f g$). When C_P or C_T is combined with other quantities such as the flight speed, due attention should be paid to the consistency of

units. Inconsistencies may be present when comparing different publications regarding the inclusion of gravity g in its definition. The difference between the use of SI units and non-SI units or a timescale in hours instead of seconds will lead to widely different numbers for SFCs. To avoid confusion, it is recommended to derive the total efficiency – this has a well-defined order of magnitude – from a given SFC by using the following relations.

- The *thermal efficiency* of a propeller engine follows from

  $$\eta_{\text{th}} = \frac{P_{\text{br}}}{FH} = \frac{1}{C_P H}\,. \tag{5.18}$$

  The BSFC of a piston engine hardly changes with speed and it is often presumed that η_{th} is constant.
- The total efficiency for jet propulsion follows from

  $$\eta_{\text{tot}} = \frac{TV}{FH} = \frac{V}{C_T H}\,. \tag{5.19}$$

  At subsonic *Mach numbers*, the TSFC of a straight turbojet does not change much, and therefore the total efficiency is roughly proportional to speed. This does not quite hold for turbofan engines, for which the approximation $C_T \approx C_1 + C_2 M$ is sometimes used, with M denoting the Mach number. The factors C_1 and C_2 depend on the altitude and the *engine rating* and can be derived from the engine manufacturer's data.

For ramjet and rocket engines the fuel or propellant burn-off is usually expressed by the *specific impulse*,

$$I_{\text{sp}} \triangleq \frac{T}{F} = \frac{\dot{m}_p\, v_j}{\dot{m}_p\, g} = \frac{v_j}{g}\,, \tag{5.20}$$

with F denoting the fuel or propellant weight-flow rate. The characteristic number I_{sp} has the dimension of time in seconds and is equal to the reciprocal value of C_T derived from the fuel consumption or propellant burning rate.

5.4 Piston engines in aviation

Single-engined light aircraft mostly have an Otto cycle engine installed. The aircraft *piston engine* has a cycle similar to that of a car engine, though it is lighter, usually air-cooled, and has a double ignition. It often also has a *supercharger* to enable high-altitude flight. Since there is a lot of experience with this type of engine, it has become very reliable and is cheaper to

purchase and maintain than a gas turbine. On the other hand, aviation gas with its high octane rating is an expensive fuel that is becoming scarce in many places. This is one of the reasons why the SI piston engine has almost completely become obsolete for applications over 300 kW.

The basic Otto cycle

In an Otto cycle engine, the air flows through the inlet to the carburettor,[15] where it is mixed with the desired amount of fuel. The pilot controls a gas valve that regulates the flow of the fuel-air mixture through the intake manifold and valves to the cylinders, where it goes through the cycle. This cycle sets the piston in a reciprocating (up-and-down) motion which is converted into crankshaft rotation by connecting rods. The crankshaft rotation – with or without the help of a step-down gearing – drives the engine shaft and the propeller.

An Otto cycle engine works with an intermittent process which means that the air in each cylinder goes through all the processes before new air is let into the cylinder, the start and end values for pressure and volume being the same. Another characteristic is that it uses an open cycle, in that the working fluid is discarded in the final expansion and replaced by a fresh charge. The cycle determines the variation in cylinder pressure as a function of the volume enclosed by the piston. The position of a piston in an Otto engine (Figure 5.13) varies between the top dead center (TDC) and the bottom dead center (BDC). The increase in pressure between these positions, known as the *pressure compression ratio*, is determined by the volumetric compression ratio. This is the ratio of the volume enclosed in the cylinder by the pistons in the BDC and the TDC, respectively.[16] The motion of the piston and the valve positions during two crankshaft rotations is described as follows:

1. On the intake stroke the piston moves from the TDC to the BDC, decreasing the pressure in the cylinder. With the intake valve open, the fuel/air mixture is induced to flow into the cylinder at a near-constant pressure.
2. During the compression stroke the valves are closed. The mixture is compressed by the upward motion of the piston to the desired burning and expansion conditions. The charge is ignited at such a moment that com-

[15] Many modern aircraft piston engines are injection engines where the fuel is injected before or inside the cylinders, instead of being mixed with air in a carburettor.

[16] The volumetric compression ratio of a spark ignition engine is typically between 8 and 10, for diesel engines it is about 15 [2].

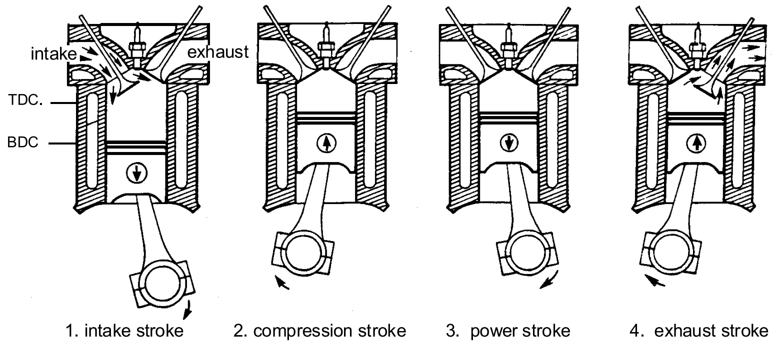


Figure 5.13 Positions of the piston and the valves during two crankshaft rotations of an Otto engine.

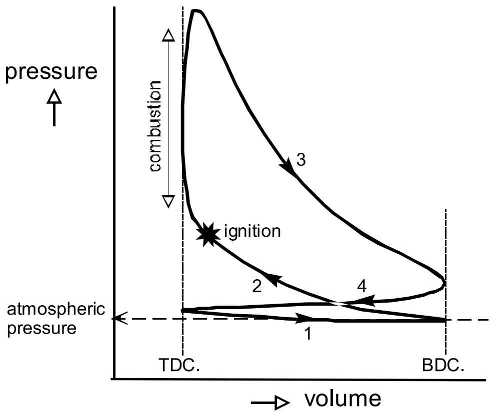


Figure 5.14 *Indicator diagram* of a naturally aspirated Otto cycle engine.

bustion is complete when the piston is just at the TDC. Combustion takes place almost instantly at constant volume.

3. The pressure that has strongly increased during combustion pushes the piston downward to the BDC while the gas expands. During the power stroke the valves are still closed, though the exhaust valve will open just before the piston reaches the BDC.
4. During the exhaust stroke the exhaust valve is open and the gas is expelled at constant pressure by the upward motion of the piston toward the TDC. The intake valve opens just before the end of the exhaust stroke.

The variation in pressure inside the cylinder can be displayed in an indicator diagram (Figure 5.14) that is determined experimentally by simultaneously measuring the piston displacement and the pressure. In principle, the power generated by the cylinder can be determined from it.

In the intermittent combustion process there is almost complete (stoichiometric) combustion. The fuel-air mixture composition that gives maximum power is approximately 1/13 to 1/15, but in an engine it can vary from 1/8 (rich mixture) to 1/20 (lean mixture). During combustion high pressures (40 to 60 bar) and high temperatures (2,700 to 3,200°C) occur. Although these temperatures only occur for short periods of time, it is necessary to cool the cylinders. Most modern aircraft piston engines are *air cooled*, although *liquid cooling* of the cylinder heads is also common. Cylinders in air-cooled engines have cooling fins that enable a good heat dissipation because of their large surface area. In liquid cooled engines water, glycol or a mixture of these substances is guided along (part of) the cylinders and then cooled in a radiator.

Supercharging

At a given mixture composition and thermal efficiency, the shaft power is proportional to the air mass drawn into the engine per unit of time and therefore to the *air density*. In a *normally aspirated engine* the intake air is guided directly into the intake manifold. Because of this the power declines linearly with the air density, which at 6,000 m altitude is only 60% of the value at sea level. Since piston-engine powered aircraft with a *pressure cabin* have their best cruise altitude between 6,000 and 7,500 m, they need engines with supercharging. The boost pressure of a supercharged engine, and with that the cylinder intake air density, is increased by a separate compressor, known as a *supercharger*.

A supercharger can be driven mechanically by the engine shaft; gearing is needed to achieve a high enough compressor speed. To prevent overloading the engine structure by an excessively high cylinder pressure, the engine is not set to full power at low altitudes. With increasing altitude, the gas valve is opened further to maintain a constant intake pressure. Due to the decreasing atmospheric temperature, the shaft power gradually increases with altitude. At the *critical altitude* the engine is at full power, at still higher altitudes the power decreases with the lower air density, similar to a normally aspirated engine. Sometimes a piston engine has two superchargers in series in order to sustain maximum power at even higher altitudes.

An important drawback of a mechanically driven supercharger is that at high altitudes the power needed to drive the supercharger significantly reduces the net shaft power. Moreover, when the engine is abruptly throttled

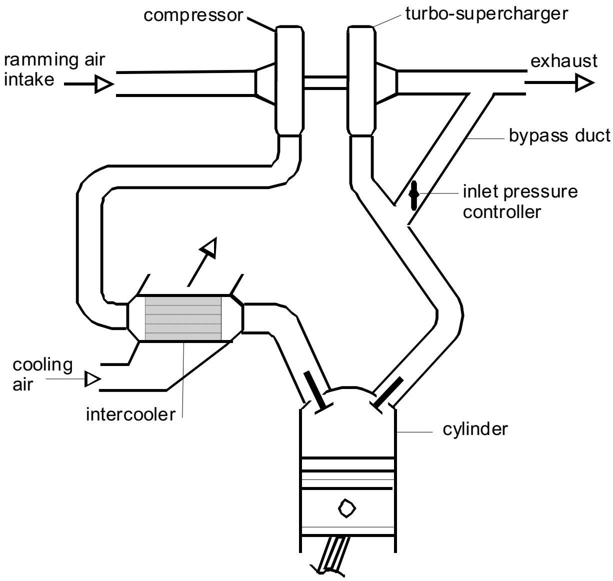


Figure 5.15 Schematic drawing of a *turbo-supercharger* with an intercooler.

down it is impossible for the fast-rotating supercharger to follow this deceleration and large forces will be exerted on the gearing, unless the supercharger is decoupled. For these reasons, the supercharger is preferentially driven by a turbine that is placed in the exhaust gasses, an *exhaust turbo-supercharger* (Figure 5.15). This system uses the energy in the exhaust gasses that otherwise would have been lost and thus in principle it does not use any shaft power. Also the maximum sea-level power can be sustained up to a higher altitude than with a mechanically driven supercharger.

The intake pressure of a turbo-supercharger is automatically set to a predefined value by means of a valve in the by-pass channel (waste gate) of the turbine. At sea level the valve is open and most of the exhaust gasses pass through the channel. With increasing altitude the valve is gradually closed and more of the gas flows through the turbine, thus increasing the supercharger power. However, this causes the intake air temperature to rise and the air mass flow to decrease. To prevent this effect from reducing the engine power, an intercooler is placed in between the supercharger and the cylinder intakes. Figure 5.16 shows the variation in engine power with altitude for a supercharged engine compared to a normally aspirated engine with the same cylinder displacement. Installing a supercharged engine enables a much bet-

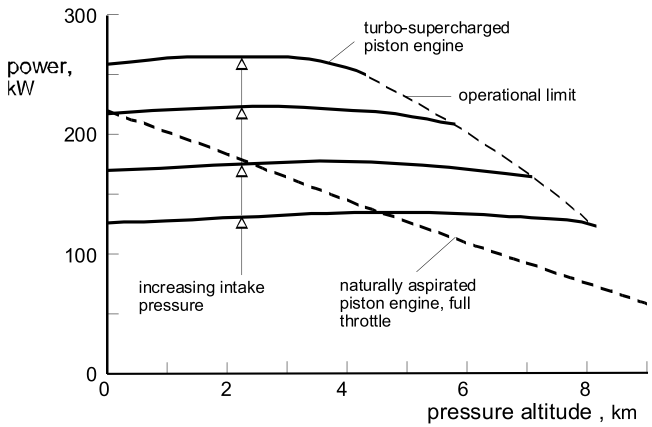


Figure 5.16 Effect on piston engine shaft power of altitude variation and supercharging.

ter performance of propeller aircraft at high altitudes and as a side-effect a turbo-supercharger works as an exhaust noise damper.

Shaft power

The power delivered to the crankshaft by each piston equals the product of the average cylinder pressure during the power stroke p_m – this is called the mean effective pressure (MEP), the piston surface area, and the piston displacement S_p. From the cycle description it follows that there is one power stroke for two crankshaft rotations. Therefore the shaft power summed over all cylinders (index c) is equal to

$$P = N_c \left(p_m \frac{\pi}{4} D_p^2 S_p \right) \frac{n_c}{2} = p_m \Delta_p \frac{n_c}{2} \; . \tag{5.21}$$

Here N_c denotes the number of cylinders, D_p the piston diameter and n_c the crankshaft rpm. The engine's swept volume Δ_p is equal to $N_c(\pi/4)D_p^2 S_p$. The compression ratio is determined completely by the cylinder geometry, independent of the shaft speed. If the ambient and intake pressures are given, the same can be said of p_m and hence the shaft power is proportional to the engine rpm. However, the flow and heat losses and mechanical friction increase at high rpm and the power no longer increases proportionally. Above a certain rpm, shaft power and thermal efficiency will decrease.

Power limits set by the engine manufacturer are controlled through the rpm, the intake boost pressure and the gas mixture.

1. At take-off a rich mixture is needed to generate maximum power. Because of the highly loaded engine, take-off power can be sustained for a few minutes only.
2. Maximum continuous power – for some engine types equal to take-off power – can be used during a few dozens minutes for generating a high performance. This may be necessary in the case of *engine failure*.
3. Maximum cruise power – usually about 75% of take-off power – is allowed for unlimited duration.
4. Recommended economical cruise power is 60 to 65% of take-off power. A lean mixture is used to obtain a low fuel consumption.

The maximum engine speed for a piston engine is typically between 2,500 and 3,000 rpm, although it can be higher for small engines. At this rotational speed a propeller would be inefficient and produce a lot of noise. Therefore most piston engines have a 0.5 to 0.7 step-down gear ratio.

5.5 Gas turbine engine components

Gas generator and energy conversion

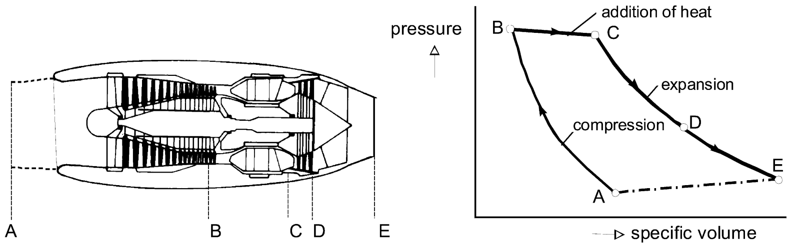
Nowadays gas turbines are used almost exclusively in commercial and military aviation. These generate heat energy by *internal combustion* of jet fuel with oxygen from the air flowing through the engine. Contrary to a piston engine, the airflow in a gas turbine engine is continuous and the successive processes take place in rotating and stationary components that have a fixed position within the engine. The engine working process is an open *Brayton cycle*[17] (Figure 5.17), consisting of the following components.

A-B Atmospheric air is drawn through the *air intake* into one or more rotating compressors which increase the pressure, for some engine types to more than 40 bar and, consequently, also the temperature and density.

B-C Vaporized fuel is injected[18] in the combustion chamber(s) and the fuel-air mixture burns at a constant (high) pressure. Because of the rise in

[17] The Englishman Brayton proposed the concept of the gas turbine cycle and was the first to built successful engine hardware (1872).

[18] Ignition of the fuel is required only when the engine is started up.



(a) Engine components (b) Brayton cycle

Figure 5.17 Components and cycle of a jet engine.

temperature, the specific volume and the velocity of the gas increase strongly.

C-D The hot gas drives one or more turbines in which the pressure, density, and temperature drop. Every turbine – this essential component has given its name to this engine type – drives a compressor by means of a stiff hollow shaft.

D-E The energy present in the exhaust gasses can be converted into propulsive power in several ways.

All aviation engine types that are part of the gas turbine family (Section 5.2) are based on the *gas generator* as described above, which is also called the *core engine*. Its operation is controlled primarily by varying the fuel flow into the combustion chamber. A larger fuel flow leads to a larger energy supply to the turbine(s), thus to a higher rotational speed of the core engine, with increased pressures and air mass flow. This leads to strongly increasing heat energy, which can be converted into mechanical energy by means of one of the following options.

- A *turbojet engine* generates thrust by expansion of the hot gasses inside the exhaust *nozzle* – and eventually also behind it – where a high speed propulsive jet is formed.
- The core of a *turbofan engine* has the same cycle as the straight *jet engine*, but it produces a jet (hot flow) with lower energy because part of the generated heat energy is extracted by a turbine and used to compress by-passing air. The low-pressure compressor needed for this is either part of the gas generator or a *fan* driven by a separate turbine. By-pass air is not involved in the combustion process and expands into a relatively cool jet (cold flow).
- In a *turboprop engine* the energy is predominantly extracted by a power turbine that drives the propeller shaft via a step-down gearing. The

thrust generated by the expanding gasses is a small but worthwhile fraction – less than 15%, depending on the flight speed – of the propeller thrust.

- A *turboshaft engine* generates only shaft power and is used mainly for powering helicopters. The gearing to reduce the rpm is normally not an integral part of the engine.

Gas turbine engines have some parts in common, those will now be discussed.

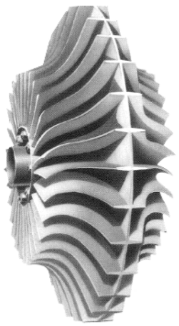
Subsonic air intake

The *air intake* duct is a carefully designed diffuser to make sure that the inlet air reaches the compressor under the desired conditions. At low flight speeds the engine sucks in atmospheric air, at high speeds a deceleration is needed to $M \approx 0.55$ at the compressor face. There should be a minimal loss of *total pressure* and *flow separation* should be avoided because compressors are sensitive to an irregular intake flow. Every intake shape is therefore a compromise and some (short) intakes use extra inlet doors which open at low speeds. Civil aircraft engine nacelles have intakes with sound-absorbing linings in the ducts. The air intakes of some supersonic aircraft have a variable geometry which makes them mechanically complicated (Section 9.9).

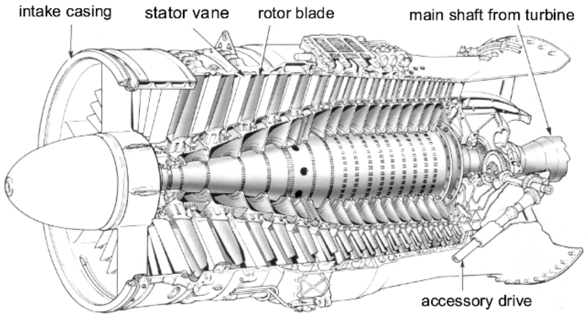
Compressor

The thermal efficiency of a gas generator improves when the combustion chamber pressure is made as high as possible, hence, the pressure increase in the *compressor* is crucial. A distinction is made between centrifugal (or radial-flow) and axial-flow compressors.

The main components of a *centrifugal compressor*, as depicted in Figure 5.18(a), are the intake casing, the impeller, the diffuser, and the outlet casing. The impeller rotates at a very high speed (20,000 to 30,000 rpm) and accelerates the air by means of the centrifugal effect. The kinetic energy is converted into pressure in the non-moving diffuser and then the air takes a 90° turn in the outlet casing toward the manifold and into the combustion chamber. The impeller speed is limited by the maximum permissible stresses in the material at the rotor perimeter and by the condition that the flow remains subsonic. This gives a maximum pressure ratio of about five for an



(a) Impeller for a centrifugal compressor



(b) Single-spool axial flow compressor

Figure 5.18 Gas turbine engine compressors (courtesy of Rolls-Royce plc).

aluminium alloy compressor and about seven for titanium. Since this is not enough for a favourable engine cycle, there is often an axial-flow compressor placed in front of the centrifugal compressor. Notwithstanding that they have a lower performance, centrifugal compressors are still used – mainly in smaller engines – because of their structural simplicity and robustness.

The central part of an *axial-flow compressor*, Figure 5.18(b), is a rotor with a large number of sequential blade rows. The rotor blades are twisted and have a thin, highly cambered section. Their aerodynamic action increase the pressure and velocity of the airflow. In between each rotor blade row, a row of stator blades attached to the compressor casing prevents the air from rotating, thereby keeping it flowing in an axial direction. The flow is forced through a continuously decreasing cross-sectional area, which increases the pressure and temperature but hardly changes the average velocity. A combination of one rotor blade row with the following row of stators is called a compressor stage. Axial-flow compressors are characterized by the number of stages, sometimes eight to 16. Part of the compressed airflow is diverted to cool the high-pressure turbine, another part is used to deliver bleed air to onboard systems like the environmental control system (ECS) for the pressure cabin.

A single axial-flow compressor delivers a larger pressure ratio (6 to 12) with higher efficiency than a centrifugal compressor. This is still not enough to obtain a total pressure ratio up to 18 to 30, say and therefore twin-spool or even triple-spool rotor systems are used. The low-pressure compressor is connected to the low-pressure turbine with a shaft, these three components together forming the low-pressure spool. The hollow shaft of the high-

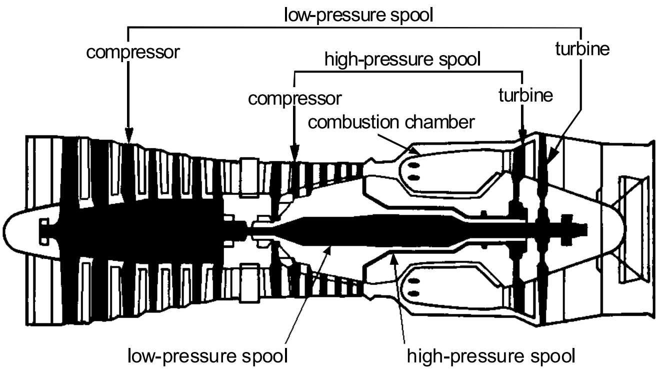


Figure 5.19 Scheme of a twin-spool turbojet engine.

pressure *compressor spool* rotates around the shaft of the high-pressure spool (Figure 5.19). Since the spools rotate at different speeds, both compressors and turbines can work at their optimum speed. Turboprops have a maximum rpm of 15,000 to 50,000 for large and small powers, respectively, for turbofans the high-pressure rotor rotates at 7,000 to 30,000 rpm.

Compared to a centrifugal compressor, the axial-flow compressor handles a four times larger air mass flow per unit of frontal area. For the same thrust, the engine is lighter and more compact, has a smaller frontal area, and can therefore be installed in a smaller space. Although their development and fabrication are costly, axial-flow compressors are used in all present-day gas turbine engines in commercial aircraft and high-performance military aircraft.

Combustion chamber

The early gas turbines had a ring of large can *combustion chambers* – Figure 5.7 shows an example – modern gas generators with axial-flow compressors have one compact annular burner, with fuel injected by swirl vanes at the front end. Staged or co-annular pairs use one as a pilot optimized for minimum pollution at low power. Combustion takes places at constant pressure, with a flame temperature of about 2,500 K. Over the years, burner exit temperatures have increased from about 1,300 K to almost 1,800 K for modern large turbofans. In normal operation the fuel/air ratio is between 1:45 and 1:130, whereas for stoichiometric combustion of hydrocarbon fuel it is

close to 1:15. So the primary combustion zone must be shrouded in cold air and only a small part of the total flow is involved in the primary combustion. About 60% of the inlet air is gradually inserted to cool the gasses, thereby ensuring that the turbine entry temperature (TET) stays within its limit.

Turbine

A *turbine* extracts mechanical energy from the hot gasses leaving the combustion chamber. This is used to drive the high-pressure compressor and also, depending on its use, a low-pressure compressor and/or a fan, a propeller, or a helicopter rotor. The turbine also powers accessories like fuel pumps, oil pumps, and an electrical generator. Modern gas turbines have two or three *compressor spools*, that is, separate combinations of a high and a low-pressure compressor and turbine, each connected by a central shaft and rotating at different speeds. Similar to compressors, turbines can be of the radial or axial flow type. Axial flow turbines are dominating in large turbofan engines, a combined axial-radial flow machinery can be found in small turbofans and turboprop engines. The turbine torque is delivered by one or more rows with curved and twisted blades, that are radially connected to the turbine disc. In front of the turbine and in between the rotor blades, radial stator blades attached to the engine case ensure a proper flow direction towards the rotor blades. Since the turbine rotates at high speed in a hot gas, its blades have to negotiate high mechanical and thermal loads. Complex (and expensive) high-temperature alloys including a high percentage of nickel are essential for turbines. In addition, cooling air diverted from the compressor is injected to circulate through the front turbine blade rows and is then ejected.

Exhaust nozzle

The gasses leaving the turbine still have a high temperature[19] and a moderately high pressure. In gas turbines for subsonic or transonic flight, they are accelerated through expansion of a convergent *exhaust nozzle* to local sonic speed at the exit. The diameter of the nozzle exit is such that in the design condition the static pressure is approximately equal to the ambient pressure. The nozzle is part of the exhaust system, which can also include a system for

[19] The turbine exhaust gas temperature (EGT) is measured and used as a parameter to control the engine operation.

thrust reversal, a mixer of hot and cold flows and sound-absorbing treatment. In military aircraft there is often an *afterburner* between the gas turbine and the exhaust, requiring a nozzle exit with variable area. The exhaust system of a high-speed aircraft is described in Section 9.9.

5.6 Non-reheated turbojet and turbofan engines

Turbojets

Non-reheated *turbojet engine* have just a short propulsive nozzle behind the turbine and, apart from rockets, this was the simplest form of *reaction propulsion*[20] between 1940 and 1960. In Section 5.2 we derived the thrust T of a jet engine from the *momentum equation*. Neglecting the small contribution of the fuel mass flow in Equation (5.2) we define the *net thrust* as follows:

$$T_{\text{net}} = \dot{m}_a(v_e - V) + (p_e - p_\infty)A_e. \tag{5.22}$$

This can be split into *gross thrust*,

$$T_{\text{gross}} = \dot{m}_a v_e + (p_e - p_\infty)A_e, \tag{5.23}$$

and the *intake momentum* (or ram) *drag*,

$$D_{\text{ram}} = \dot{m}_a V. \tag{5.24}$$

The net thrust can thus be written as

$$T_{\text{net}} = T_{\text{gross}} - D_{\text{ram}}. \tag{5.25}$$

Due to the high jet velocity, the gross thrust is much larger than the intake momentum drag and the net thrust falls off only slightly with the flight speed, provided this is low. Not only is the magnitude of the thrust important, but also its centre of action. A characteristic pressure, temperature, and gas velocity variation is given in Figure 5.20.

All engine and nacelle components that are exposed to internal flow contribute to the thrust. As an example, static ($V = 0$) thrust components from pressure forces are given in Figure 5.21. The net thrust is the (relatively small) resultant of various large forces, noticeably the forward forces on the

[20] Although the Concorde was developed after 1960, its installation as a whole is much more complex than that of subsonic civil aircraft at that time (Section 9.9).

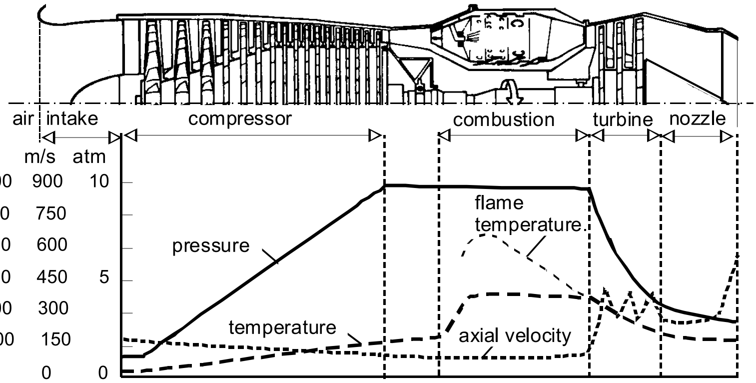


Figure 5.20 Variation of pressure, temperature and flow velocity in a jet engine (courtesy of Rolls-Royce plc).

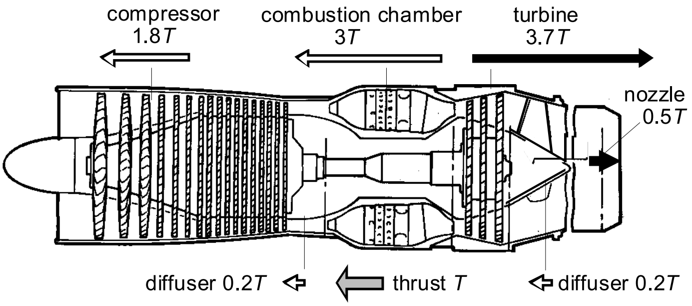
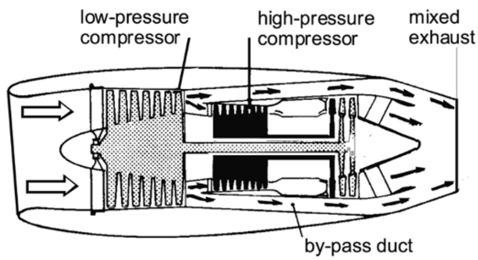


Figure 5.21 Distribution of internal thrust components for a static turbojet.

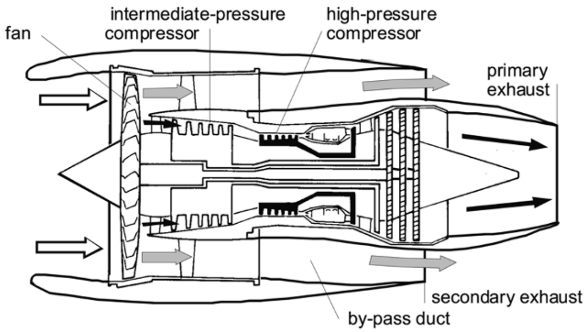
combustion chamber and the compressor and the backward forces on the turbine and the exhaust pipe. The opposing forces lead to large internal loads on the engine structure. In high-speed flight, intake air is compressed in the inlet diffuser that feels a forward pressure force it. Moreover, part of the thrust works on the external exposed area of the nacelle as a nose *suction force*.

Turbofans

Modern fighters and civil jet aircraft have *turbofan engines*. These are based on the principle of the dual-flow engine, but with some significant differences in design and use (Figure 5.22).



(a) Low by-pass ratio turbofan



(b) Three-shaft high by-pass ratio turbofan

Figure 5.22 Main turbofan engine configurations (courtesy of Rolls-Royce plc).

(a) *By-pass engines* have a low *by-pass ratio*. The intake air first passes through a low-pressure compressor with several stages which increase the pressure with a ratio between two and four. The inner 50 to 80% of the intake air flows through the high-pressure compressor of the *core engine*, hence the by-pass ratio is between 0.2 and 1.0. The outer flow by-passes the core engine and is usually mixed with the hot flow to slightly increase the efficiency. By-pass engines are used only in older types of (noisy) fighters.

(b) The intake air of a high-bypass turbofan is first compressed by a *fan* at the front of the engine (front fan).[21] This is usually a single-rotor compressor with a pressure ratio of about 1.5, made of titanium or a carbon fiber reinforced synthetic material, followed by a row of *stator blades*. Because of the large fan diameter, its blade tips may reach a helical speed

[21] In principle it is possible to place the fan on the circumference of the turbine that drives it. Such a "rear fan" is nowadays an outdated configuration, but may reappear in future engines; see Figure 5.39.

of up to 500 m/s during take-off, which requires very thin and locally swept aerofoils. The primary flow, some 10 to 20% of the intake air, flows through the core engine and leaves the primary nozzle as a hot flow. The much larger secondary cold flow by-passes the core through a duct and expands in a secondary nozzle, or is mixed with the hot flow before leaving a common nozzle. This type of engine is used primarily in transonic airliners and executive aircraft.

The thrust of a turbofan engine is composed of contributions from the primary and secondary flows; for separated exhaust flows both can be determined with Equation (5.22). This equation can also be used for a mixed exhaust flow if the gas velocity is considered to be an average value determined by the mixing process.

The dual-flow principle leads to a reduced average jet velocity and a higher *propulsive efficiency*. Due to high pressure in the combustion chamber, the *thermal efficiency* is also higher. As a result of both improvements, the fuel consumption is significantly lower. In short, a turbofan engine has a higher efficiency than a straight turbojet engine with the same thrust because thrust is generated with a larger engine airflow ejected at a lower velocity. This is achieved for a larger intake diameter, at the expense of a larger nacelle weight and drag. Turbofan engines are more complex and more expensive per unit of thrust to buy and to maintain compared to straight turbojet engines. For civil application, the decisive factor is the considerable fuel consumption reduction.

By-pass ratio

An important characteristic of a turbofan engine is its *by-pass ratio*,

$$B \triangleq \frac{\dot{m}_{\mathrm{c}}}{\dot{m}_{\mathrm{h}}} \, . \tag{5.26}$$

The indices (h for hot and c for cold) refer to the primary, respectively the secondary, flow. The by-pass ratio depends on the operational conditions and it is usually specified for the static condition at sea level. Neglecting the pressure component, the net thrust of a turbofan engine with separate exhaust flows is

$$T_{\mathrm{net}} = \dot{m}_{\mathrm{h}}(v_{j,\mathrm{h}} - V) + \dot{m}_{\mathrm{c}}(v_{j,\mathrm{c}} - V) \, . \tag{5.27}$$

When the fuel mass flow is neglected the total engine mass flow rate is $\dot{m}_a = \dot{m}_{\mathrm{h}} + \dot{m}_{\mathrm{c}}$ and Equation (5.26) can be used to derive the by-pass ratio,

$$\frac{\dot{m}_{\mathrm{h}}}{\dot{m}_a} = \frac{1}{1+B} \quad \text{and} \quad \frac{\dot{m}_{\mathrm{c}}}{\dot{m}_a} = \frac{B}{1+B}. \tag{5.28}$$

Combination of these equations yields the thrust

$$T_{\mathrm{net}} = \dot{m}_a \left(\frac{v_{j,\mathrm{h}} + B v_{j,\mathrm{c}}}{1+B} - V \right). \tag{5.29}$$

For the hypothetical case that $v_j = v_{j,\mathrm{h}} = v_{j,\mathrm{c}}$ we find the earlier derived equation (5.3) for the *ideal thrust*,

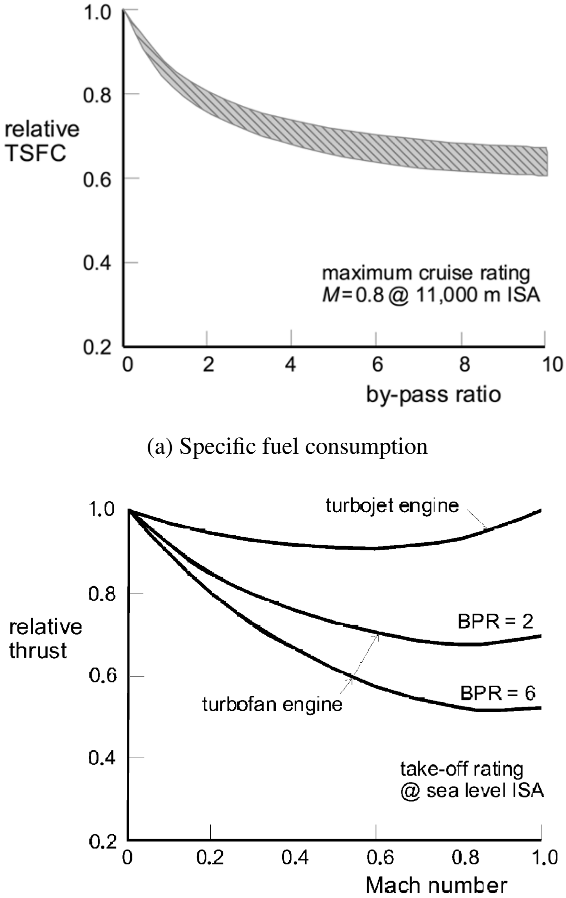
$$T_{\mathrm{net}} = \dot{m}_a (v_j - V). \tag{5.30}$$

The *total efficiency* appears to have a maximum value for $v_{j,\mathrm{c}}/v_{j,\mathrm{h}} \approx$ 0.75 to 0.85. When B is increased, the optimum fan pressure ratio, the jet velocities $v_{j,\mathrm{h}}$ and $v_{j,\mathrm{c}}$ and the *specific thrust* $T_{\mathrm{net}}/\dot{m}_a$ decrease. Since for fighter aircraft a high specific thrust is important, high-bypass engines are not installed. In commercial aircraft, on the other hand, high efficiency is of paramount importance. This explains the differences in the design of both types of turbofans, depending on their application.

Figure 5.23(a) illustrates how sensitive the *specific fuel consumption* is to the by-pass ratio. For instance, for $B = 5$ the SFC is 70% of that of a turbojet engine ($B = 0$) with the same *gas generator* cycle. However, turbofans operate at higher pressures and temperatures, which gives it a higher thermal efficiency and makes the difference even more impressive. Figure 5.23(b) shows the downside: the thrust decay with flight speed is influenced negatively when the by-pass ratio is higher. On the other hand, it can also be stated that – for the same thrust at cruise conditions – a turbofan has a higher static thrust when the by-pass ratio is higher. Modern airliners with high BPR turbofans have therefore better take-off performance than the older generations with turbojets or low-bypass engines.

As of the 1960s turbofans have been developed with a by-pass ratio of four to nine and an overall pressure ratio[22] of 25 to 40. A typical example is the IAE V-2500 (Figure 5.24) that was developed by an international consortium of engine manufacturers. It is installed in the Airbus A318/319/320/321, among others. The IAE V-2500 and the R-R Trent, see Figure 5.22(b), are triple-spool engines, with three turbines to drive the fan, the intermediate-pressure compressor and the high-pressure compressor, respectively, through three concentric rotor shafts. The total efficiency at cruise conditions of these engines is of the order of 35%.

[22] The overall pressure ratio (OPR) is the ratio between the combustion chamber inlet pressure and the engine intake pressure.



(a) Specific fuel consumption

(b) Thrust lapse with speed

Figure 5.23 Influence of the *by-pass ratio* on specific fuel consumption and take-off thrust of turbofans.

Engine noise

The high jet velocity of a pure jet engine causes an objectionable sound level of 110–120 PNdB[23], measured on the ground during take-off. Jet noise can be reduced by mixing the jet and the external flow and the undamped noise produced by turbofans decreases when the by-pass ratio is increased (Figure 5.25). However, fan-generated noise dominates when the by-pass ratio

[23] PNdB means perceived noise level in decibels. Based on the sensitivity of the human ear, this number is derived from the frequency distribution of the sound pressure spectrum.

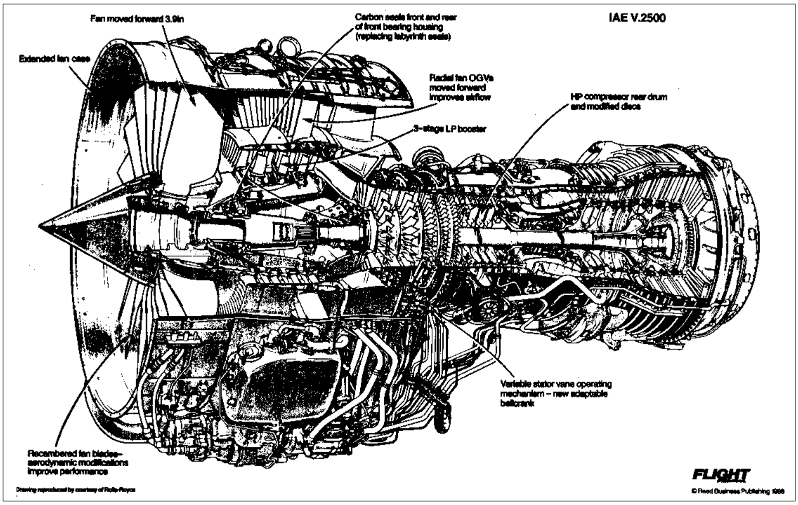


Figure 5.24 The IAE V-2500 *turbofan engine* with a by-pass ratio of 5.4; static take-off thrust: 111 kN. (Courtesy of *Flight International*)

is more than about 1.5. Since its source is inside the nacelle, it is possible to reduce fan noise with acoustic treatment, such as noise-absorbing lining attached to the intake and by-pass ducts. *Engine noise* can be reduced by mixing the primary and secondary exhaust flows, which may also reduce fuel consumption. With increasing power absorbed by the fan, its diameter and thus the blade tip speed also increase. This effect can be delayed by operating the turbine at a lower rotational speed, but for a by-pass ratio of nine or higher, the turbine would rotate too slowly to be effective. Alternatively, the fan rotational speed can be reduced by placing a reduction gear between the core engine and the fan. This leads to the geared engine (GTF), which is under development at the time of writing.

5.7 Turboprop and turboshaft engines

Configurations

In a *turboprop engine*, the low-pressure turbine extracts gas energy generated by the core engine and delivers the power through gearing – usually at the front part of the engine – to the propeller shaft. Ten to 15% of the available gas energy is converted in the nozzle into kinetic energy so that the

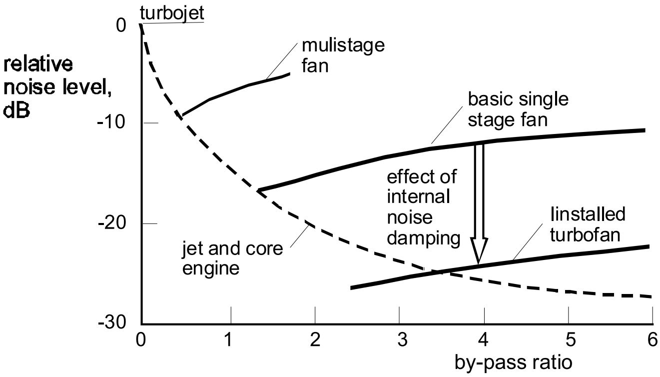
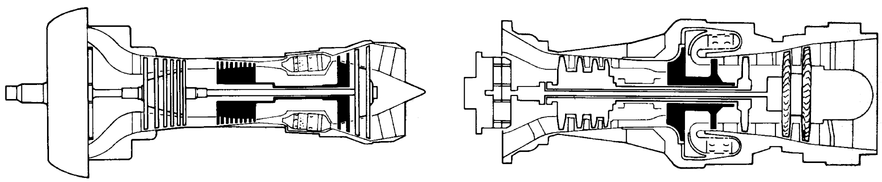


Figure 5.25 Influence of by-pass ratio and acoustic treatment on engine noise level (source: [20], modified).



(a) Twin-spool axial flow turboprop engine (b) Twin-spool turboshaft engine with free power turbine

Figure 5.26 Mechanical arrangement of gas turbine engines driving a propeller or a helicopter rotor (courtesy of Rolls-Royce plc).

engine also delivers some thrust. This contribution to the power available for propulsion amounts to 5 to 10% and diminishes with increasing flight speed. Turboprop engines can be classified according to the method of transferring turbine power to the propeller shaft.

- If the propeller shaft is driven by the low pressure *compressor spool*, the propeller speed is proportional to the gas generator (low pressure) rotational speed. Figure 5.26(a) shows an example.
- If the propeller shaft is driven by a separate *free power turbine*, its speed can be controlled independently from the gas generator operation. The engine responds more rapidly to variations in the fuel flow.

Most turboprop engines have twin-spool rotors. The compressors can be centrifugal and/or axial flow types, giving a *total pressure* ratio between 12 and 20. Figure 5.27 shows an example of a turboprop engine with two centrifugal compressors.

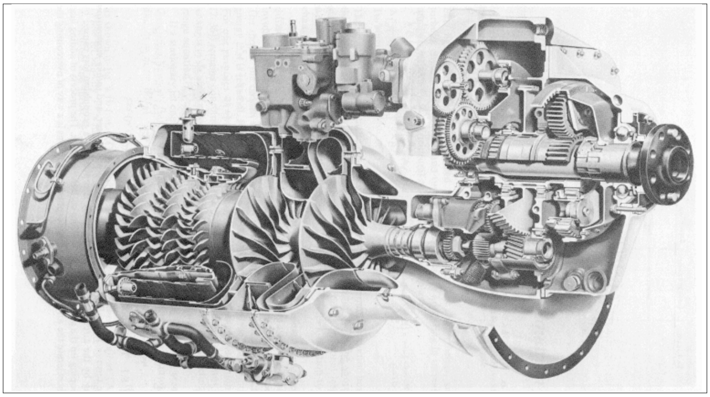


Figure 5.27 The Garret TPE-331-14 1,227 kW single-shaft turboprop engine.

In *turboshaft engines* all available energy is extracted and converted into shaft power by a separate (free power) turbine, as in Figure 5.26(b). The engine gasses leave the engine at a low velocity through a wide *exhaust nozzle*. In aviation, this type of engine is typically used in helicopters where the shaft is connected to a transmission gear which drives the main rotor. A special class of turboshaft engines is employed in auxiliary power units (APU) to serve as a backup power source in the air and to deliver pneumatic and electric power on the ground to the on-board systems. This renders the aircraft independent from ground support equipment.

Power and fuel consumption.

In a turboprop engine with shaft power $P_{\rm br}$ the exhaust gasses produce a jet thrust T_j,

$$T_j = \dot{m}_a(v_j - V). \tag{5.31}$$

The total *available power* for a *propeller efficiency* η_p is

$$P_{\rm av} = \eta_p P_{\rm br} + T_j V. \tag{5.32}$$

The *equivalent power* is defined as the (fictitious) shaft power for which the available propeller power would be equal to the combination of the actual shaft power and thrust,

$$\eta_p P_{\rm eq} = \eta_p P_{\rm br} + T_j V, \quad \rightarrow \quad P_{\rm eq} = P_{\rm br} + T_j V/\eta_p \,. \tag{5.33}$$

Under static conditions the thrust contribution is about 10% of the thrust. The *specific fuel consumption* is referred to the shaft power,

$$C_{P_{\mathrm{br}}} = \frac{F}{P_{\mathrm{br}}}\,, \tag{5.34}$$

or to the equivalent power,

$$C_{P_{\mathrm{eq}}} = \frac{F}{P_{\mathrm{eq}}}\,. \tag{5.35}$$

Different from turbofans, these characteristics are hardly affected by the flight speed. The SFC values cannot be compared to those for jet propulsion because their definitions and dimensions are different. The thermal efficiency according to Equation (5.12) – for a modern turboprop this can be more than 0.35 – can be used instead. For a propeller efficiency of 0.85, the total efficiency is approximately 0.30 for Mach numbers up to $M \approx 0.60$. This is lower than the total efficiency of a high BPR turbofan at $M \approx 0.80$.

Turboprop and piston engines compared

When the processes inside a gas turbine engine and a piston engine are compared, the following fundamental differences can be observed.

- In an Otto cycle engine the processes take place intermittently in four consecutive piston strokes in each cylinder which experiences, as do other engine components, a variable load. The fuel-air mixture is ignited at the end of each compression stroke. In a gas turbine engine, on the other hand, there are separate components for compression, combustion and expansion. These processes are continuous, the loads do not fluctuate significantly and the gas turbine is almost free of vibrations. The vaporized fuel needs to be ignited only for starting up the engine. A drawback is its slower reaction to instant variations in fuel flow, which means the engine (thrust) responds slowly to throttle variations.
- For any piston engine operating condition, its *compression ratio* is based on cylinder geometry and, hence, there is little variation in cylinder pressure and thermal efficiency. In fact, mechanical friction losses increase the piston engine BSFC at high engine speed. In a gas turbine the overall compressor pressure ratio depends strongly on the (fuel flow controlled) high-pressure rotor speed. With increasing speed, the thermal efficiency increases and the specific fuel consumption decreases until an optimum

value is obtained. To achieve high efficiency, a gas turbine must be operated at a high throttle setting.

- In a gas turbine engine, combustion takes place at a constant pressure and with an excess amount of air and thus the temperature does not rise as much as in the piston engine cylinders. Because the maximum pressure in a gas turbine is not that high, relatively light structures can be used and the combustion chamber temperature obviates the need for the expensive high octane fuel. However, the thermal efficiency is lower and the fuel consumption per unit of power is higher.

An Otto engine produces little power per litre of cylinder displacement, mainly because only one out of four strokes does all the work. The engine structure is heavy and the need to cool the cylinders entails considerable mechanical complication and air drag. The gas turbine, on the other hand, is compact and has a high power production per unit of frontal area, although the high turbine speed requires a higher gear ratio towards the propeller or rotor shaft. For the same power, a turboprop engine weighs only about a third of a piston engine, and its nacelle is smaller and lighter. In combination with the much higher reliability and cheaper fuel of the gas turbine, its advantages extensively compensate for the higher price and fuel consumption.

5.8 Gas turbine engine operation

Engine control

The pilot *controls the engine* fuel flow from the cockpit by setting the position of a power lever, which in turn makes an input to the engine control system. This imposes limitations at low altitudes and high rotational and flight speeds to prevent overloading of the engine structure. The power lever position selects a thrust level from idle to maximum and the control system manipulates the control variables, while observing the operating engine limits. Apart from *thrust reversal*, engine performance is controlled by only one parameter: the high-pressure spool speed or the engine pressure ratio (EPR). This is defined as the average total pressure[24] in the exhaust nozzle divided by the compressor *intake* total pressure. For a given control setting, gas turbine performance depends on the atmospheric temperature and pressure and

[24] Total or *stagnation pressure* is the combined effect of static and *dynamic pressure*.

the flight Mach number. *Engine operation* is monitored at the flight deck through one or more of the following indications:

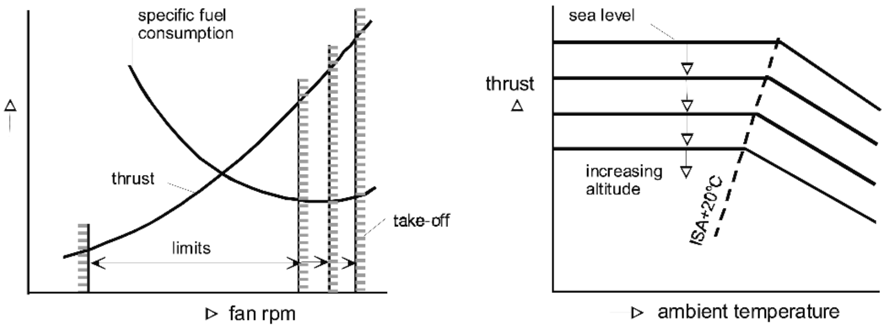
- The fuel supply, characterized by the flow rate $\dot{m}_f$. The selection of tanks and the monitoring of the remaining fuel is derived from this.
- Engine shaft speed, that is, the rpm of the fan and the gas generator rotor spools. Since for high by-pass ratio turbofans most of the thrust is delivered by the fan, its rpm forms a good measure. Engine acceleration is limited to avoid compressor surge.
- The turbine entry temperature (TET), an indicator for the thermal load on the engine which has a large influence on the lifetime. This very high temperature cannot be measured directly and is therefore derived from the exhaust gas temperature (EGT).
- The shaft torque of a turboprop or turboshaft engine, derived from the measured oil pressure in the gear box. The shaft power at a given rpm is proportional to this pressure.

The thrust or power level that a gas turbine engine can generate for a certain period of time is limited by *engine ratings*. For turbofan engines the following ratings should be observed to achieve the desired lifespan and maintenance cost limits:

- *Take-off thrust:* the maximum thrust allowed during take-off, which can only be used for 5 minutes at most.
- *Climb thrust:* the maximum thrust allowed during the climb to cruise altitude.
- *Maximum continuous thrust:* the maximum thrust allowed without a time limit that can be used for the operating engines if one engine is inoperative.
- *Cruise thrust:* the maximum thrust allowed in high speed *cruising flight*.

Turboprop engines have comparable limits on the shaft power and/or torque.

The first generations of gas turbines were controlled by hydromechanical systems, modern-day gas turbines avail of a full authority digital engine control (FADEC) system. This enables a more complex logical system to be used, resulting in better performance and improved reliability.



(a) Effect of rotational speed (b) Effect of temperature

Figure 5.28 Turbofan static thrust and TSFC affected by engine speed and ambient temperature.

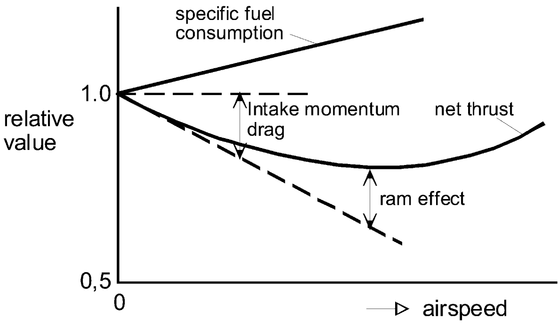
Operational conditions

Temperature

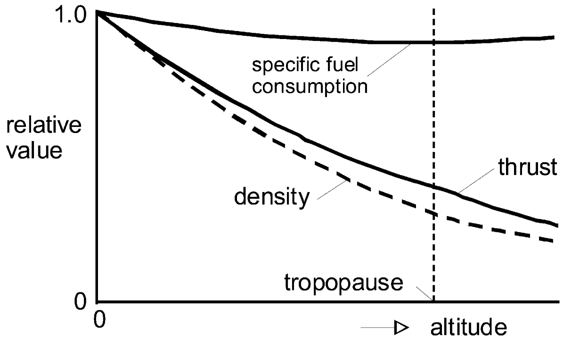
The effects of engine rotational speed and ambient conditions will be discussed and illustrated qualitatively for a turbofan engine under a static condition at sea level (Figure 5.28). These variations are essential for aircraft performance analysis as treated in Chapter 7.

(a) When the engine power lever is moved to the take-off position, engine speed, mass flow, internal temperatures, and pressures will increase and thrust increases progressively with rpm. The TSFC decreases initially until it levels off and starts to increase, the fuel-economical operating condition occurring at a high rpm. Low fuel consumption is hardly relevant for taking off, but gas turbine engines show similar behaviour at cruise conditions. The figure also indicates a minimum engine speed, for which it functions properly during ground and flight idling.

(b) For a high outside temperature, the *air density* is low and engine thrust decays. This is undesirable for taking off in a hot climate and therefore more fuel is admitted for increasing the internal temperatures and pressures. At low and normal outside temperatures, the thrust is constrained by the permissible loading of the engine structure up to a certain limit: the engine is “flat rated”. In the example the maximum outside temperature for this flat rating is 35°C – at sea level this is 20°C above the standard value – and above this limit the take-off thrust decreases appreciably.



(a) Effect of flight speed at constant altitude



(b) Effect of altitude at constant airspeed

Figure 5.29 Turbofan thrust and TSFC affected by flight speed and altitude.

Flight speed

Turbofan thrust is determined by Equation (5.22), stating that the net thrust is equal to the gross thrust reduced by the *intake momentum drag*; see Figure 5.29(a). For a constant air mass flow rate $\dot{m}_a$, nozzle exit velocity v_e, and pressure p_e, the *gross thrust* is constant, whereas the intake momentum drag increases proportional to the speed. Neglecting the pressure term in Equation (5.22), the *ideal (net) thrust* lapse amounts to

$$\frac{T}{T_{V=0}} = \frac{v_j - V}{v_j} = 1 - \frac{V}{v_j} < 1\,. \tag{5.36}$$

Relative to the static condition, the thrust lapse is significant when the jet has a low velocity v_j, which is the case for high by-pass engines; see Figure 5.23(b). However, at high flight speeds the engine total intake pressure,

the exit pressure, and velocity will increase. This is known as the *ram compression* effect, which leads to increasing gross thrust. This implies that the initially decreasing net thrust levels off and then increases above a certain Mach number. For straight turbojets, the thrust becomes even larger than the static thrust ($T/T_{V=0} > 1$), for high by-pass turbofans the ram effect becomes manifest mainly at high subsonic speeds. At high altitudes the gross thrust increment and the intake momentum drag term more or less cancel each other out. The net thrust does not vary a great deal, though it is much lower than the static thrust ($T/T_{V=0} < 1$). The TSFC of a straight *jet engine* does not depend much on the flight speed, Figure 5.29(a), for turbofans the variation is larger.

Up to $M \approx 0.5$ the flight speed hardly influences the shaft power of a turboprop engine. However, the conversion of shaft power into propeller thrust varies strongly. In contrast with jet propulsion, turboprop BSFC decreases slightly with increasing speed. This is caused mainly by the difference in the definition of C_T (for jet propulsion) and C_P (for propeller propulsion).

Altitude

When a turbofan aircraft climbs to cruise altitude, the ambient pressure, density, and temperature gradually decrease until the *tropopause* is reached. The decreasing density causes the air mass flow rate to decrease and thus the thrust is declining. Figure 5.29(b) shows that the thrust lapse is not proportional to the density in the *troposphere*, a result of the decreasing temperature. The *thermal efficiency* increases and the TSFC decreases slightly. Since the stratospheric temperature is constant, the thrust is (theoretically) proportional to the ambient pressure. The TSFC rises slowly with altitude because the Reynolds number drops. The behaviour as sketched in Figure 5.29(b) suggests that the ideal operating altitude for a turbofan engine is roughly at the tropopause.

Turboprop engine power decreases with the altitude approximately proportional to the ambient density, although at low altitudes a limit on the engine torque can lead to a constant available shaft power. When the necessary cruise power is taken as the reference, an aircraft with turboprop engines needs more take-off power than an aircraft with supercharged piston engines, whose power will only drop off above a rather high *critical altitude*. As with turbofan engines, the BSFC decreases sensibly with increasing altitude in the troposphere.

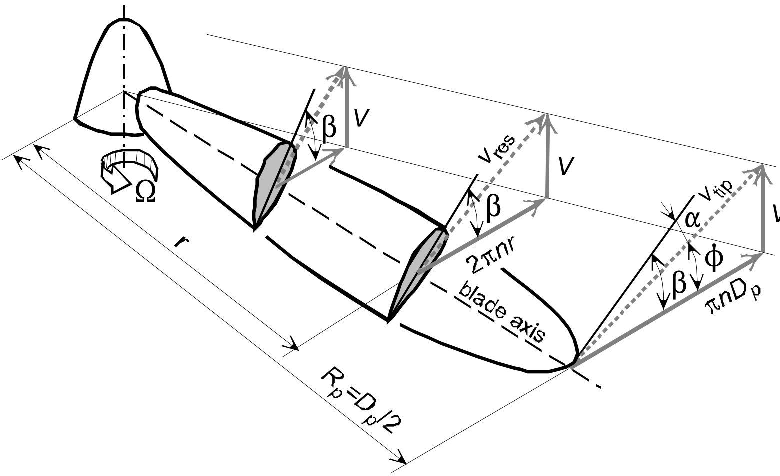


Figure 5.30 Variation of angles and velocities along a propeller blade.

5.9 Propeller performance

Propeller blade geometry

At low subsonic flight speeds the *propeller* is the pre-eminent device to convert shaft power into propulsive power. A propeller consists of two to six *propeller blades* which are often simultaneously adjustable around a blade axis in the *propeller plane* perpendicular to the shaft. The mechanism to adjust the blades is installed in a *streamlined* hub around the propeller shaft. The shape of a propeller blade resembles that of a wing; especially their *aerofoil sections* are very similar. Both have been designed to create an aerodynamic lift with as little drag loss due to viscosity and vorticity as possible. Contrary to a wing, a propeller blade is highly twisted because of the propeller rotation.

Propeller blades have a variation of *twist*, *chord* and *thickness* from hub to tip (Figure 5.30). For a blade section at radius r, the *blade pitch* β denotes the angle between the chord and the propeller plane. Given a rotational speed n, the angular speed around the propeller shaft is $\Omega = 2\pi n$. The resulting speed v_{res} of the blade section is composed of the translational speed V in the direction of the propeller shaft and the peripheral speed Ωr due to the rotation in the propeller plane.[25] The *advance angle* ϕ is measured between

[25] It has been assumed here that the propeller plane is normal to the direction of flight. Induced velocities which will be discussed later are neglected here.

the resulting velocity and the propeller plane, hence

$$\tan\phi = \frac{V}{\Omega r} = \frac{V}{2\pi n r} \,. \tag{5.37}$$

This angle is smaller if the blade section is further away from the propeller shaft. The *angle of attack* α is measured between the section chord and the resulting velocity,

$$\alpha = \beta - \phi \,. \tag{5.38}$$

This angle determines the ratio of the sectional lift-to-drag ratio, which should have a maximum value in one or more design conditions, such as take-off climb or cruising flight. For optimum propeller performance, the blade section angle of attack should be approximately the same for every radius. In order to match the advance angle variation, the blade pitch is made to decrease between the blade hub and the tip by twisting the propeller blades. This is also clear from Figure 5.30, showing that for a given (optimum) angle of attack α_{opt} the blade pitch is the smallest at the tip, where the radius is equal to the propeller radius R_p. For example, let us assume a propeller blade to have $\beta = 45°$ for $r = 0.20R_p$, $\beta = 15°$ for $r = 0.75R_p$ and $\beta = 11°$ for $r = R_p$. The blade is then *twisted* over an angle of $-34°$ between $r = 0.20R_p$ and the tip. The mean value of the blade pitch is usually characterized by its value at $r = 0.75R_p$, denoted $\beta_{0.75} = 15°$ in this case. For a given variation along the radius of the blade chord, the section shape and the pitch, the geometry and incidence of the propeller blade is completely defined. This information is the starting point for determining propeller performance.

Actuator disc theory

Basic equations

The Rankine–Froude *actuator disc* model introduced in Section 5.2 does not only provide insight into some basic properties, it may also be used to provide input for more elaborate flow models. The concept of Figure 5.9(a) is shown again in Figure 5.31, now with the flow surrounding the *slipstream* included. The flow through the stream tube captured by the disc, referred to as the *slipstream*, is supposed to be frictionless and incompressible. Uniform flow properties like velocity and pressure exist across any plane normal to the stream tube. The jump in pressure across the disc is discontinuous, but

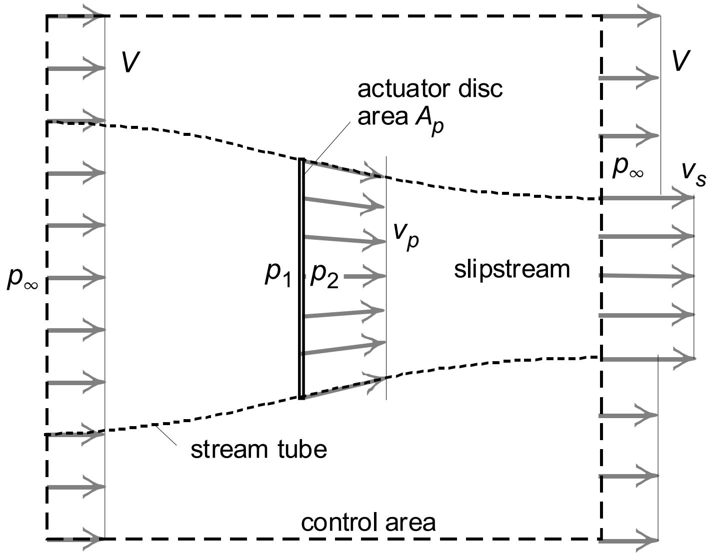


Figure 5.31 Model of the flow through a propeller according to the *actuator disc* concept.

uniformly distributed in its plane and only the axial acceleration of the slipstream is taken into account. The *control area* for applying the *momentum equation* has cylinder-shaped side faces, the front face is at infinity upstream and the aft face is in the fully developed slipstream. There is an inward airflow through the side faces. However, this reduces to zero if they are at an infinite distance from the disc. As mentioned in Section 5.2, the surrounding flow does not experience a change in momentum. It thus follows that, along their tubular boundary, the propeller and the surrounding flows do not exert a resultant force on each other ($R_x = 0$), although the pressures do vary. The propeller *thrust* is then determined solely by the change in the slipstream momentum rate,

$$T = \dot{m}_a(v_s - V) = \rho v_p A_p (v_s - V), \tag{5.39}$$

where v_p and A_p denote the velocity in the propeller plane and the disc area, respectively. The thrust also equals the force on the disc surface by the pressure jump,

$$T = (p_2 - p_1)A_p \ , \tag{5.40}$$

with p_1 and p_2 denoting the pressure just in front of and aft of the disc, respectively. These can be found by applying Bernoulli's equation (Section 3.3) to the *stream tubes* extending from the disc forward and backward

until the pressure is equal to the ambient pressure p_∞,

$$p_1 = p_\infty + \frac{1}{2}\rho\,(V^2 - v_p^2) \qquad \text{and} \qquad p_2 = p_\infty + \frac{1}{2}\rho\,(v_s^2 - v_p^2)\,. \quad (5.41)$$

Then the pressure difference is

$$p_2 - p_1 = \frac{1}{2}\rho\,(v_s^2 - V^2)\,. \qquad (5.42)$$

It is noted that, since the disc imparts energy to the flow, *Bernoulli's equation* cannot be used for the flow through it. Substituting the pressure difference from Equation (5.42) into (5.40) and equating this with (5.39) yields the solution

$$v_p = \frac{v_s + V}{2} \qquad \text{or} \qquad v_p - V = \frac{1}{2}(v_s - V)\,. \qquad (5.43)$$

Translating this simple but important result to a propeller, it states that the velocity through the propeller plane is equal to the mean value of the velocities far in front of and behind the propeller. In other words, one-half of the increase in velocity of the fully developed slipstream has already occurred at the propeller plane.

Jet efficiency

In the framework of the disc theory, the *propeller efficiency* can be determined from the power delivered by the propeller and the slipstream velocity, as follows. The *jet power* P_j added to the flow is equal to the increment of its kinetic energy per unit time,

$$P_j = \frac{1}{2}\dot{m}_a(v_s^2 - V^2) = \dot{m}_a(v_s - V)\frac{v_s + V}{2} = T v_p. \qquad (5.44)$$

This shows that the jet power is equal to the product of the thrust and the velocity in the propeller plane, where it is produced. The *available power* P_{av} is the product of the thrust and the velocity of the *undisturbed flow*,

$$P_{\text{av}} = TV. \qquad (5.45)$$

The difference between the available and the jet power is the *induced power* P_i, which is equal to the thrust times the velocity increment in the propeller plane. This power is not used for propulsion and therefore it is a loss that can be expressed by the *jet efficiency*,

$$\eta_j = \frac{P_{\text{av}}}{P_j} = 1 - \frac{P_i}{P_j} = \frac{TV}{Tv_p} = \frac{V}{v_p} = \frac{2V}{V + v_s} = \frac{2}{1 + v_s/V} \,. \tag{5.46}$$

As could be anticipated, the jet efficiency equals the *propulsive efficiency* derived in Section 5.2.

Static thrust

The thrust of a propeller is derived from its shaft power. In the simplified model of the *actuator disc*, this is jet power only and thrust is equal to $T = \eta_j P_j / V$. The variation of jet efficiency with speed according to Equation (5.46), depicted in Figure 5.10, can be used to solve the thrust. It follows that the solution for the static thrust becomes indeterminate since V and η_j are then both equal to zero. This difficulty is avoided by introducing the following dimensionless *thrust coefficient*:

$$T_C \overset{\wedge}{=} \frac{T}{q_\infty A_p} = \frac{T}{\frac{1}{2}\rho V^2 A_p} \,. \tag{5.47}$$

From Equations (5.39) and (5.43) it follows that

$$T_C = \frac{2v_p\,(v_s - V)}{V^2} = \left(\frac{v_s}{V}\right)^2 - 1 \,. \tag{5.48}$$

The velocity in the slipstream can be written as

$$v_s = V\sqrt{1 + T_C} \,, \tag{5.49}$$

and the jet efficiency is

$$\eta_j = \frac{2}{1 + \sqrt{1 + T_C}} \,. \tag{5.50}$$

From Equation (5.49) it follows that

$$q_s = q_\infty (v_s/V)^2 = q_\infty (1 + T_C) = q_\infty + T/A_p. \tag{5.51}$$

The ratio between the thrust of a propeller and its surface is called the propeller *disc loading* T/A_p. It is thus found that the dynamic pressure increment of the fully expanded slipstream equals the propeller disc loading. Application of Equations (5.39), (5.43) and (5.44) shows that, for the static condition, we have

$$v_s = \sqrt{\frac{2T}{\rho A_p}} \quad \text{and} \quad P_j = T v_p = \frac{T}{2}\sqrt{\frac{2T}{\rho A_p}}\,. \tag{5.52}$$

This yields the following relation between the static thrust and the jet power:

$$T = \left(2P_j^2 \rho A_p\right)^{1/3} \quad \text{or} \quad \frac{T}{2\rho A_p} = \left(\frac{P_j}{2\rho A_p}\right)^{2/3}. \tag{5.53}$$

This result is obviously an idealization of the actual static thrust of a propeller, which is at least 10% lower due to additional losses.

Blade-element theory

The results as derived above do not take into account the propeller blade profile drag, thrust losses at the blade tips, and slipstream swirl. Although the momentum theory gives useful information on the way propellers work, especially the change in the *momentum flow* and the kinetic energy of the flow through the propeller, it does not indicate how the thrust is exerted on the propeller. Therefore this theory is not suited for a more detailed analysis of a propeller with the intention to improve its design or to compute detailed performances. The *blade-element theory*, which calculates the aerodynamic forces on a number of blade elements and then integrates them along the propeller blade, offers this information.[26] It is more accurate because it accounts for blade-profile drag and slipstream swirl, that is, vortices and rotation of the slipstream. Some basic principles of this theory will be explained, and some important propeller coefficients introduced.

The flow around a blade element is considered in the plane parallel to the propeller shaft, at a distance r from the shaft (Figure 5.32). Propeller blade-element theory relies on the assumption that there is no flow in the radial direction (two-dimensional flow). Different from Figure 5.30, the vectors show the relative fluid velocity at the element. The resultant of the free stream velocity V is parallel to the propeller shaft and the element's peripheral velocity $\Omega r = 2\pi n r$ is

$$\begin{aligned} v_{\text{res}} &= \sqrt{V^2 + (\Omega r)^2} = V\sqrt{1 + (2\pi n R_p/V)^2 (r/R_p)^2} \\ &= V\sqrt{1 + (\pi/J)^2 (r/R_p)^2}. \end{aligned} \tag{5.54}$$

[26] According to F.E.C. Culick (1979), the Wright brothers were obliged to develop their own propeller design method, using their test results on airfoils. This formed the basis of what later became known as blade-element theory.

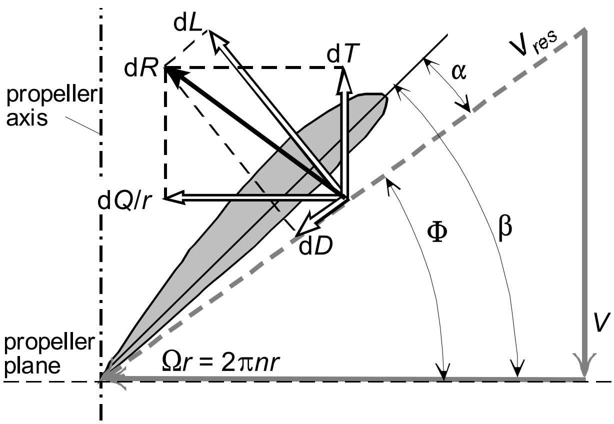


Figure 5.32 Simplified view of the forces on a *propeller blade* element.

The propeller coefficient named *advance ratio* is introduced here,

$$J \triangleq \frac{V}{nD_p} = \frac{V}{2nR_p} , \tag{5.55}$$

defining the *advance angle* of the helical path followed by the blade tips. The blade element with chord c is located between radius r and $r + \mathrm{d}r$ and the air force $\mathrm{d}R$ acting on it is resolved according to different approaches. The components $\mathrm{d}L$ perpendicular to the flow and $\mathrm{d}D$ in the direction of the flow are determined by the local conditions,

$$\mathrm{d}L = c_l \frac{1}{2}\rho v_{\mathrm{res}}^2 c \,\mathrm{d}r \qquad \text{and} \qquad \mathrm{d}D = c_d \frac{1}{2}\rho v_{\mathrm{res}}^2 c \,\mathrm{d}r . \tag{5.56}$$

Alternatively, $\mathrm{d}R$ is resolved into the thrust component $\mathrm{d}T$ in the direction of the propeller shaft and the *torque* component $\mathrm{d}Q/r$ perpendicular to it,

$$\mathrm{d}T = \mathrm{d}L \cos\phi - \mathrm{d}D \sin\phi \qquad \text{and} \qquad \mathrm{d}Q/r = \mathrm{d}L \sin\phi + \mathrm{d}D \cos\phi. \tag{5.57}$$

The blade thrust is found by integrating over all elements along the blade radius, given that the propeller hub with diameter R_h does not contribute any thrust. Multiplying by the number of blades B_p yields the propeller thrust

$$T = B_p \int_{R_h}^{R_p} (c_l \cos\phi - c_d \sin\phi) \frac{1}{2}\rho \, v_{\mathrm{res}}^2 c \,\mathrm{d}r . \tag{5.58}$$

The *thrust coefficient* C_T is defined as

$$C_T \triangleq \frac{T}{\rho n^2 D_p^4} \,. \tag{5.59}$$

For simplification, it is assumed that $R_h = 0$, hence

$$C_T = \frac{1}{8} B_p \int_0^1 (c_l \cos\phi - c_d \sin\phi) \left\{ J^2 + \pi^2 (r/R_p)^2 \right\} (c/R_p) \, \mathrm{d}(r/R_p). \tag{5.60}$$

Similarly, the propeller power P_p is found by integrating the torque components along the blade. By introducing of the *power coefficient*

$$C_P \triangleq \frac{P_p}{\rho n^3 D_p^5} \,, \tag{5.61}$$

it follows that

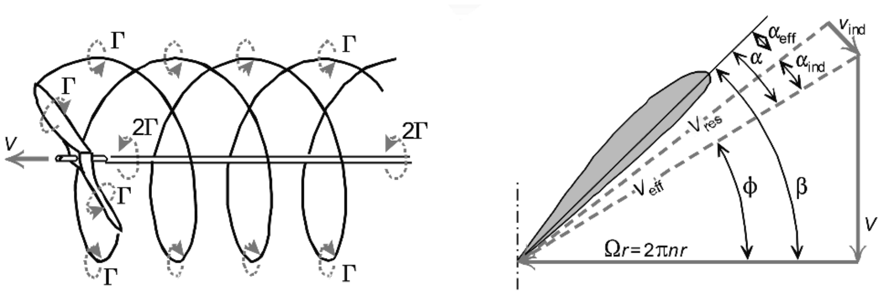
$$C_P = \frac{\pi}{8} B_p \int_0^1 (c_l \sin\phi + c_d \cos\phi) \{ J^2 + \pi^2 (r/R_p)^2 \} (c/R_p)(r/R_p) \, \mathrm{d}(r/R_p). \tag{5.62}$$

Equations (5.60) and (5.62) show that the propeller coefficients C_T and C_P are dependent variables, since both coefficients c_l and c_d are determined by the local angle of attack,

$$\alpha = \beta - \phi = \beta - \arctan(V/\Omega r) = \beta - \arctan\{(J/\pi)(R_p/r)\} \,. \tag{5.63}$$

For a given blade geometry, the variation of the relative chord c/R_p as a function of r/R_p is known and the *blade pitch* β at each radius r/R_p is determined by the *pitch angle* $\beta_{0.75}$ and the variation of twist along the blade. For a given number of blades B_p, Equations (5.60) and (5.62) indicate that the propeller coefficients are dependent on only two variables: the blade pitch and the advance ratio.

In this derivation it has not been taken into account that the relative flow around a propeller blade is not completely determined by the components V and Ωr. Analogous to the *trailing vortices* behind a wing, there is a *vortex field* behind the propeller caused by the bound vortices on the propeller blades (Figure 5.33a). This changes the flow velocity on a blade section due to the induced velocity v_{ind} and the induced angle of attack α_{ind} (Figure 5.33b). This leads to an effective flow velocity v_{eff} at an incidence $\alpha_{\text{eff}} = \alpha - \alpha_{\text{ind}}$. When this correction factor is used, there is the problem that both v_{eff} and c_l – and hence c_d – are determined by the magnitude and direction of the induced velocity v_{ind}, that itself can only be calculated when the distribution of the air forces along the blade radius are known. One feasible



(a) Vortex field behind a two-bladed propeller (b) Local flow at a blade element

Figure 5.33 Schematic visualization of vortices behind the propeller and the induced flow around a blade element.

solution for this problem is to combine the blade element and the actuator disc theories, the latter determining the axial component of v_{ind} from the momentum theory. Similarly, the tangential component of the induced flow is obtained from the angular momentum theory. A correction for the finite number of blades is also possible, as are refinements based on calculating the induced velocities from the vortex field behind the propeller. These extensions of the propeller theory will not be pursued here; summaries can be found in [2] and [7], amongst others. These refined theories still indicate that the coefficients C_T and C_P are only dependent on the blade pitch and the advance ratio, providing that there are no compressibility effects.

Propeller performance in practice

Propeller efficiency

The most important characteristic number of a propeller is its *efficiency*,

$$\eta_p \triangleq \frac{P_{\mathrm{av}}}{P_{\mathrm{br}}} = \frac{TV}{P_{\mathrm{br}}} \, . \tag{5.64}$$

In steady flight, the engine shaft power P_{br} is equal to the power absorbed by the propeller P_p. Substituting C_T from Equation (5.59) and C_P from (5.61) gives

$$\eta_p = \frac{C_T}{C_P} J. \tag{5.65}$$

This shows that the propeller efficiency only depends on the blade pitch and the advance ratio. Figure 5.34 shows a sketch – for given propeller diameter,

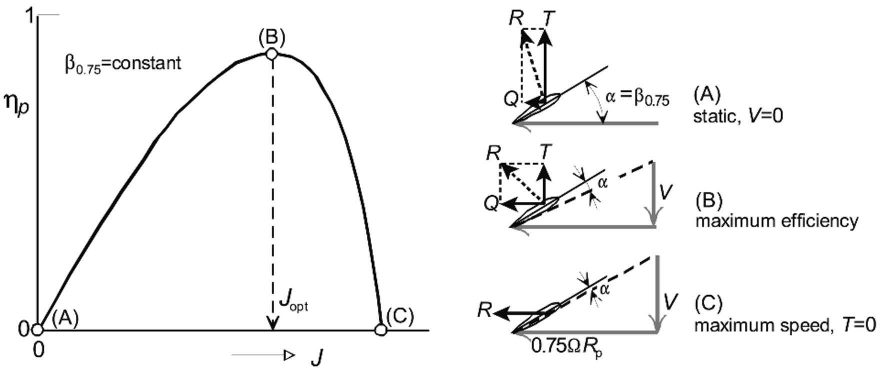


Figure 5.34 *Propeller efficiency* and *advance ratio* for various flight speeds. The blade pitch is given.

blade pitch, engine power and rpm – of the propeller efficiency as a function of the advance ratio and thus the flight speed. The following explanation is given:

(a) For the static condition, the propeller has zero efficiency but it generates a large thrust, unless the propeller blades are *stalled*.
(b) According to Equation (5.64), the propeller efficiency is proportional to the speed if the thrust were constant. Actually, the thrust decreases and the efficiency increases less than proportional to the speed. It reaches a maximum when the effective incidence of the propeller blades is optimal.
(c) If the speed increases further, the blade incidence becomes smaller and at a certain speed the blade force R is in the propeller plane, perpendicular to the direction of flight. The propeller will not generate any thrust and its efficiency is zero. There is, however, still a *propeller torque* and power is still needed.

In case of a complete *engine failure*, the torque disappears and the propeller decelerates, which causes the blade incidence to decline and then to become negative. The aerodynamic force R changes its direction and the propeller works as a windmill, keeping the engine shaft rotating. This causes a negative thrust, that is, a (considerable) windmilling drag. To avoid the windmilling condition, most propellers have a *feathering position*, where it is set non-rotating at a blade pitch of approximately 90°.

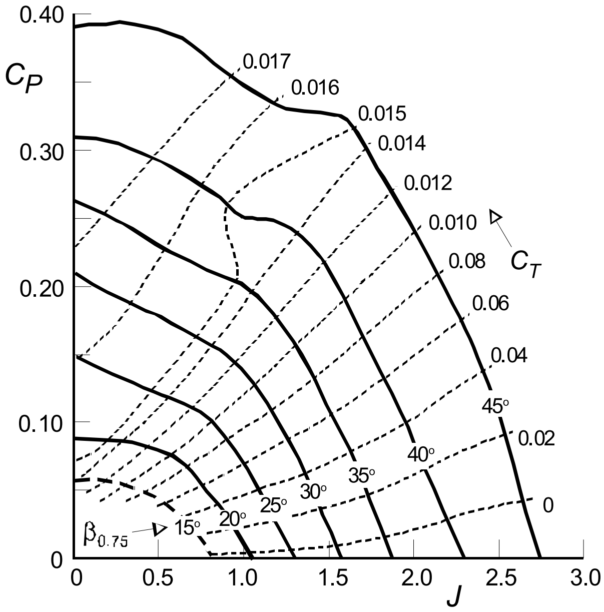


Figure 5.35 Typical *propeller diagram.*

Propeller diagrams

To describe the performance of a propeller type, propeller manufacturers issue *propeller diagrams* based on computation and measurements. These specify the coefficients C_T, C_P and η_p as functions of the advance ratio J and the blade pitch $\beta_{0.75}$. Figure 5.35 depicts an example of a propeller diagram, with $C_P = f(J, \beta_{0.75})$ and curves of constant C_T. From Equation (5.65) it follows that also $\eta_p = f(J, \beta_{0.75})$. Therefore, the propeller performance in the whole operating range can be calculated for different blade angles. Because such a diagram is, within certain limits, independent from the propeller diameter, it can be used for calculating the performance of a given propeller as well as for selecting a propeller diameter and blade pitch. Depending on their purpose, propeller diagrams have different coordinate axes. For instance, in Figure 5.36 the same propeller characteristics have been plotted as in Figure 5.35 for several values of the blade pitch.

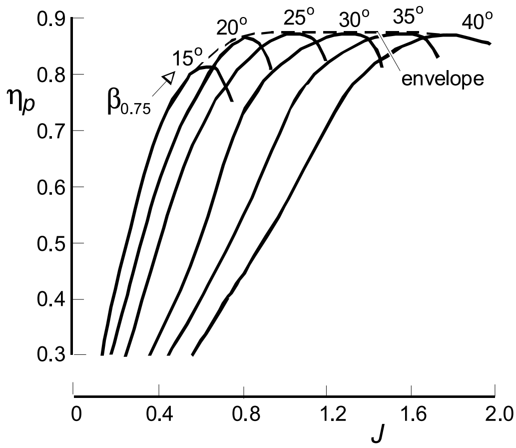


Figure 5.36 *Propeller efficiency* for a light single-engined aircraft.

Blade pitch control

Until around 1930, aircraft were equipped with one-piece wooden propellers with a built-in *blade pitch* that could not be changed. *Fixed-pitch propellers* are still being used in present-day light aircraft because of their simplicity. By looking at Figure 5.36 it becomes clear that with a fixed blade pitch a high propeller efficiency can only be achieved in a small range of advance ratios – deviating significantly from this range leads to degraded efficiencies. Figure 5.34 also shows that at high speeds, like in a *dive*, the propeller blade incidence becomes very small. The torque decreases and the propeller will rotate faster, which may lead to over-speeding of the engine. This can be avoided by using a coarse blade pitch, but this has the drawback that at take-off or low flight speeds the *propeller torque* is too large for the engine to work at full power. With an *adjustable-pitch propeller* the blade angle can be changed on the ground in anticipation of a particular flight situation. While the propeller is not rotating, the pilot may adjust it to a fine pitch for best take-off and climb performance, or to a coarse pitch for cruising.

A far more effective method is to continually adjust the blade pitch to the desired angle, depending on the engine rpm and flight speed. Figure 5.36 shows that a large range of high efficiency conditions can be obtained in this way. The envelope of the individual efficiency curves offers a high efficiency that varies little with speed. This situation is approximated by the *constant-speed propeller* that has a hub with a built-in hydraulic or electric blade adjustment. For a piston engine with a constant-speed propeller, the

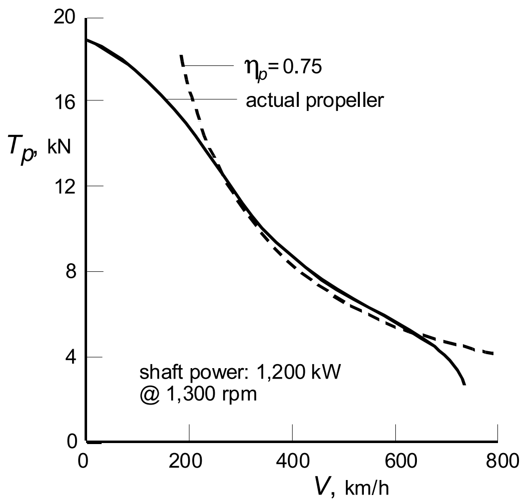


Figure 5.37 Variation with flight speed (constant altitude) of the thrust produced by a *constant-speed propeller*.

pilot avails of two independent controls: the engine rpm and the manifold inlet pressure, through the fuel throttle.[27] For a turboprop engine, the fuel supply, the engine, and the propeller rpm are all coupled and the pilot has only one control. Turboprops with a *free power turbine* may have the additional possibility of controlling the blade pitch from the flight deck. In this case, the gas generator speed and power are automatically adjusted to keep propeller speed constant. Such a beta control system makes it possible to promptly vary the power and thrust at a constant rotational speed of both the engine and the propeller. This eases the control of the angle of descent and manoeuvring on the ground.

The influence of the flight speed on the thrust of a constant-speed propeller can be determined from Equation (5.64):

$$T = \eta_p \frac{P_{\rm br}}{V} \, . \tag{5.66}$$

For a given speed, the propeller efficiency is read from the propeller diagram, for which the engine shaft rpm is used to calculate the coefficients J and C_P (Figure 5.37). Equation (5.66) states that the thrust for a constant propeller efficiency is inversely proportional to the velocity. For $V = 0$ this would

[27] Power and fuel consumption are also affected by the fuel/air mixture ratio and the operation of a supercharger. Engine control may thus seem to be a difficult task, but nowadays automation has improved this.

lead to an infinite static thrust, but since in reality the propeller efficiency is also zero the result of Equation (5.66) is undetermined. Equation (5.53) may be used to find a first approximation for the static thrust, propeller data must be obtained from the manufacturer for higher accuracy. At high velocities there may be losses at the blade tips because of compressibility, as will be discussed hereafter. In conclusion, the assumption of a hyperbolic variation of the thrust may therefore be acceptable for medium speeds as it creates significant errors at low and high speeds.

High tip speeds

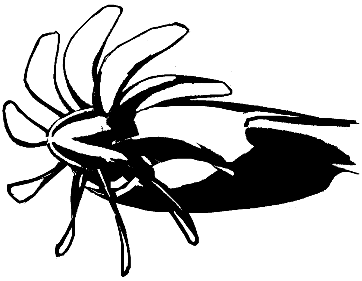
The helical speed of the blade tips can be derived from Figure 5.30:

$$v_{\text{tip}} = \sqrt{V^2 + (\pi n D_p)^2} \quad \rightarrow \quad M_{\text{tip}} = M\sqrt{1 + (\pi/J)^2}\,, \qquad (5.67)$$

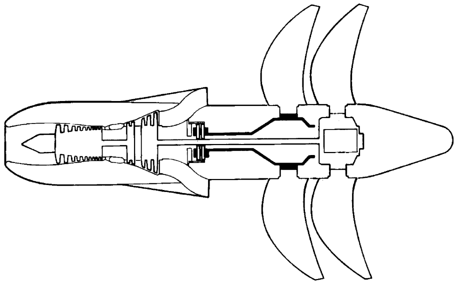
with M denoting the flight *Mach number*. The tip speed is much larger than the flight speed and at high flight and/or rotational speed it may exceed the speed of sound. For instance, for $J = 2.0$ and $M > 0.54$ the blade tips experience a supersonic flow velocity ($M_{\text{tip}} > 1$). The accompanying shock waves strongly reduce the efficiency and increase propeller noise. Up to the present day, the maximum flight speed of propeller aircraft has therefore been limited to $M \approx 0.65$. High propeller tip speeds should also be avoided during taking off and climb-out as, with a high rpm, the propeller may generate an objectionable noise level.

As long as compressibility is not an issue, the *total efficiency* of propeller propulsion can be higher than that of jet propulsion. After the energy crisis in the 1970s, propeller and engine manufacturers developed the *propfan*,[28] as depicted in Figure 5.38(a), for flight Mach numbers up to 0.80. Compared with a conventional propeller, a propfan has a large number (8 to 12) of blades and a high disc loading P/D_p^2, that is, an unusually small propeller diameter compared to the power generated. Its crescent-shaped blades are very thin, with sweptback tips. The first propfans had a single blade row, which improved the propeller efficiency at $M = 0.75$ by 15%. Research in the 1980s showed that two contra-rotating blade rows, as in Figure 5.38(b), achieve an even better performance by avoiding slipstream swirl. Their additional 5% improvement makes the propeller efficiency comparable to that of a conventional propeller at low subsonic speeds.

[28] A version developed and tested by General Electric was called *un-ducted fan* (UDF). Also the terms *open rotor* and *high-speed propeller* are used.



(a) Gas turbine-powered propfan proposed by Hamilton Standard (1975)



(b) Propfan concept with contra-rotating propellers (courtesy of Rolls-Royce, plc)

Figure 5.38 Concepts of propellers for high-subsonic speeds.

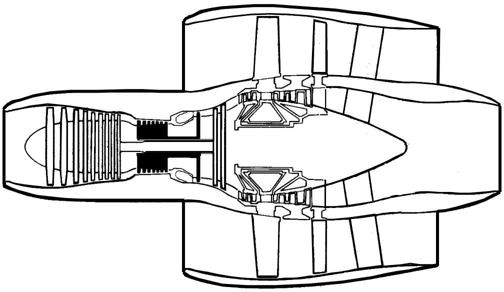


Figure 5.39 Concept of an ultra-high by-pass turbofan with contra-rotating blade rows (courtesy of Rolls-Royce plc).

Although flight tests have confirmed the high performance level of propfans, development and installation problems and the relatively high noise level during the take-off have prohibited their use up to the present time. Several engine manufacturers and laboratories therefore investigate the merits of *ducted fans* for high Mach numbers, also called advanced ducted propellers (ADP) or superfans (Figure 5.39). Combined with gas turbines, these are considered as *turbofan engines* with a very high *by-pass ratio*. The use of such engines might bridge the gap between turbofans and turboprops, which would also make the distinction between propeller and jet propulsion less relevant. Apart from the Ivchenko D-27 engine, there are no propfans and ducted fans in production at the present time. The increasing emphasis on the environmental issue and fossil fuel shortages may change this in the not too distant future.

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